Buckling Analysis of Woven Glass epoxy Laminated Composite Plate

A Thesis Submitted In Partial Fulfillment
of the Requirements for the degree of

Master of Technology
In
Civil Engineering
(Structural Engineering)

By
ARUN KUMAR R
Roll No-207CE208

Department of Civil Engineering
National Institute of Technology Rourkela
Rourkela-769008,
Orissa, India

May, 2009

Buckling Analysis of Woven Glass epoxy Laminated Composite Plate

A Thesis Submitted In Partial Fulfillment
of the Requirements for the degree of

Master of Technology
In
Civil Engineering
(Structural Engineering)

By
ARUN KUMAR R
Roll No-207CE208

Under The Guidance of
Prof. S. K. Sahu
and
Prof. A. V. Asha

Department of Civil Engineering
National Institute of Technology Rourkela
Rourkela-769008,
Orissa, India

May, 2009

NATIONAL INSTITUTE OF TECHNOLOGY
ROURKELA – 769008, ORISSA
INDIA

CERTIFICATE

This is to certify that the thesis entitled, "**BUCKLING ANALYSIS OF GLASS EPOXY LAMINATED COMPOSITE PLATES"** submitted by Mr**. Arun Kumar R** in partial fulfillment of the requirement for the award of **Master of Technology** Degree in **Civil Engineering** with specialization in **Structural Engineering** at the National Institute of Technology, Rourkela (Deemed University) is an authentic work carried out by him under my supervision and guidance.

To the best of my knowledge, the matter embodied in the thesis has not been submitted to any other University/ Institute for the award of any degree or diploma.

Date: May 30, 2009

Place: Rourkela

Prof. S. K. Sahu

Prof. A. V. Asha

Dept of Civil Engineering

National Institute of Technology

Rourkela – 769008

Acknowledgments

I would like to express my gratitude to my guide, **Dr. S K Sahu** and **Prof. A V Asha**, for their encouragement, advice, mentoring and research support throughout my studies. Their technical and editorial advice was essential for the completion of this dissertation. Their ability to teach, depth of knowledge and ability to achieve perfection will always be my inspiration.

My sincere thanks to **Dr. S. K. Sarangi,** Director and **Prof M. Panda,** Head of the Civil Engineering Department, National Institute of Technology Rourkela, for his advice and providing necessary facility for my work.

I am very thankful to all the faculty members and staffs of civil engineering department who assisted me in my research, as well as in my post graduate studies. I would also like to thank Prof. B. B. Verma and other supporting staff in the Metallurgical & Materials engineering for their help.

I also thank all my batch mates, who have directly or indirectly helped me in my project work and in the completion of this report. I also thank to Shuvranshu, Ravi, and all first year students for their friendly environment in civil computer laboratory.

Finally, I am grateful to my parents K.G Radhakrishnan Nair and Suchetha. V for their love, support and guidance. They have always been supportive of my academic pursuit.

Arun Kumar R

CONTENTS

ABSTRACT

There have been numerous studies on the composite laminated structures which find many applications in many engineering fields namely aerospace, biomedical, civil, marine and mechanical engineering because of their ease of handling, good mechanical properties and low fabrication cost. They also possess excellent damage tolerance and impact resistance. The mechanical behaviour of composite structures is of particular interest to engineers in modern technology. Buckling of plates is a well-established branch of research in composite structures stability. It has a wide range of applications in engineering science and technology. Buckling behaviour of laminated composite plates subjected to in-plane loads is an important consideration in the preliminary design of aircraft and launch vehicle components. The sizing of many structural subcomponents of these vehicles is often determined by stability constraints. Plates with circular holes and other openings are extensively used as structural members in aircraft design. The buckling behaviour of such plates has always received much attention by researchers. These holes can be access holes, holes for hardware to pass through, or in the case of fuselage, windows and doors. In some cases holes are used to reduce the weight of the structure. In aerospace and many other applications these structural components are also made up of composite material to further reduce the weight of the structure. The outstanding mechanical properties of composite structures, such as durability and corrosion-resistance characteristics combined with low density, make it more attractive compared to conventional materials.

In this study, the influence of cut-out shape, length/thickness ratio, and ply orientation and aspect ratio on the buckling of woven glass epoxy laminated composite plate is examined experimentally. Clamped –free -Clamped-free boundary condition is considered for all case. Experiments have been carried out on laminated composites with circular, square and rectangular cut-outs. The thickness of the plate was changed by increasing the number of layers. After the buckling experiments micro electroscopic scanning was performed for the failed specimens. Comparisons are made between the test results, by using two different approach. The results shows effect of various cut-out shapes, orientation of fiber, aspect ratio and length to thickness ratio on the buckling load.

LIST OF FIGURES

LIST OF TABLES

CHAPTER-1

INTRODUCTION

INTRODUCTION

In many engineering structures such as columns, beams, or plates, their failure develops not only from excessive stresses but also from buckling. Only rectangular thin plates are considered in the present study. When a flat plate is subjected to low in-plane compressive loads, it remains flat and is in equilibrium condition. As the magnitude of the in-plane compressive load increases, however, the equilibrium configuration of the plate is eventually changed to a non-flat configuration and the plate becomes unstable. The magnitude of the compressive load at which the plate becomes unstable is called the “critical buckling load.”

A composite material consist of two or more materials and offers a significant weight saving in structures in view of its high strength to weight and high stiffness to weight ratios. Further, in a fibrous composite, the mechanical properties can be varied as required by suitably orienting the fibres. In such material the fibres are the main load bearing members, and the matrix, which has low modulus and high elongation, provides the necessary flexibility and also keeps the fibres in position and protect them from the environment.

Development of new applications and new composites is accelerating due to the requirement of materials with unusual combination of properties that cannot be met by conventional monolithic materials. Actually, composite materials are capable of covering this requirement in all means because of their heterogeneous nature. Properties of composite arise as a function of its constituent materials, their distribution and the interaction among them and as a result an unusual combination of material properties can be obtained .

Laminated composites are gaining wider use in mechanical and aerospace applications due to their high specific stiffness and high specific strength. Fiber-reinforced composites are used extensively in the form of relatively thin plate, and consequently the load carrying capability of composite plate against buckling has been intensively considered by researchers under various loading and boundary conditions. Due to the excellent stiffness and weight characteristics, composites have been receiving more attention from engineers, scientists, and designers. During operation the composite laminate plates are commonly subjected to compression loads that may cause buckling if overloaded. Hence their buckling behaviours are important factors in safe and reliable design of these structures.

In view of difficulty of theoretical and numerical analysis for laminated structure behaviours, experimental methods have become important in solving the buckling problem of laminated composite plates. This work deals with buckling analysis of symmetrically and laminated composite plates under clamped -free –clamped- free boundary condition. The effects on buckling load by cut out size, length/thickness ratio, ply orientation, and length/breadth ratio are investigated.

Review of literature

Fiber-reinforced composites are used extensively in the form of relatively thin plate, and consequently the load carrying capability of composite plate against buckling has been intensively considered by researchers under various loading and boundary conditions. Thus far, there have been numerous studies on the fabric woven composite laminated structures which find widespread applications in many engineering fields namely aerospace, biomedical, civil, marine and mechanical engineering because of their ease of handling, good mechanical properties and low fabrication cost. They also possess excellent damage tolerance and impact resistance.

The initial theoretical research into elastic flexural- torsional buckling was preceded by Euler’s (1759) treatise on column flexural buckling, which gave the first analytical method of predicting the reduced strengths of slender columns, and by Saint-Venant’s 1855 memoir on uniform torsion, which gave the first reliable description of the twisting response of members to torsion. However, it was not until 1899 that the first treatments were published of flexural-torsional buckling by Michell and Prandtl, who considered the lateral buckling of beams of narrow rectangular cross-section. Their work was extended by Timoshenko to include the effects of warping torsion in I-section beams. Most recently the invention of high-speed electronic computers exerted a considerable influence on the static and dynamic analysis of plates.

Chen and Bert (1976) investigated optimal design of simply supported rectangular plates laminated to composite material and subjected to uniaxial compressive loading. Numerical results are presented for optimal-design plates laminated of glass/epoxy, boron/epoxy, and carbon/epoxy composite materials.

Linear elastic buckling of plates that are subjected to in-plane forces is a problem of great practical importance that has been extensively researched over the past 60 years. Elastic instability of flat rectangular plates became an important research area when the design of the lightweight airframes was introduced. Fok (1984), has been applied the theory of thin plates to engineering structures. Some advantages of thin-walled structures are high strength coupled with the ease of manufacturing and the relative low weight. However, thin-walled structures have the characteristic of susceptibility of failure by instability or buckling. It is therefore important to the design engineer that accurate methods are available to determine the critical buckling strength.

Laminated plates with strip-type delamination under pure bending were investigated analytically and experimentally by Yeh and Fang (1997). In the analysis, a two dimensional nonlinear finite element code based on updated lagrangian formulation was developed to analyze the bending behaviour of the laminated plates and the local buckling phenomenon of the sub laminates in the delaminated region. The formulation includes large displacements and large rotations needed to describe the local buckling phenomenon of the delaminated region.

Radu and Chattopadhyay (2000) used a refined higher order shear deformation theory to investigate the dynamic instability associated with composite plates with delamination that are subject to dynamic compressive loads. Both transverse shear and rotary inertia effects are taken into account. The theory is capable of modelling the independent displacement field above and below the delamination. All stress free boundary conditions at free surfaces as well as delamination interfaces are satisfied by this theory. The procedure is implemented using the finite element method.

Hwang and Mao (2001) conducted the non-linear buckling and post-buckling analyses to predict the delamination buckling load and delamination growth load. In order to predict the delamination growth load, the total strain-energy release rate criterion, criterion of strain-energy release rate component, and inter laminar-stress criterion are used. Experimental results are also provided to compare with the prediction.

A procedure for determining the buckling load of the aluminium rectangular plate is presented by Supasak and Singhatanadgid(2002) .Buckling load of aluminum rectangular plates are determined using four different techniques, i.e. (1) a plot of applied load vs. out-of-plane displacement, (2) a plot of applied load vs. end shortening, (3) a plot of applied load vs. average in-plane strain, and(4)the Southwell plot. In this study, buckling loads determined from different experiment methods were compared with the theoretical buckling loads.

A dynamic analysis model is proposed by Wen-pei and Lin Cheng (2003) to acquire buckling load of plate. We used the dynamic measured data from selected test points and by modal analysis got the modal parameters-mode shape and frequency; and then, derived a flexible matrix with the above model parameters. Force analysis was used to get the flexible matrix of equivalent force and the characteristic equation for determining the buckling load of the member.

. Wang and Lu (2003) was carried out an investigation to understand the buckling behaviour of local delamination near the surface of fiber reinforced laminated plates under mechanical and thermal loads. The shape of the delaminated region considered is rectangular and triangular. The displacement expression is composed of items with the effect of tension-shear coupling and the effect of bend-torsion coupling. The critical strains of laminated plates with various shaped local delamination and different stacking patterns are obtained by making use of the energy principle.

Shukla and Kreuzer (2005) proposed a formulation based on the first-order shear deformation theory and von-Karman-type nonlinearity to estimates the critical/buckling loads of laminated composite rectangular plates under in-plane uniaxial and biaxial loadings. Different combinations of simply supported, clamped and free boundary conditions are considered. The effects of plate aspect ratio, lamination scheme, number of layers and material properties on the critical loads are studied.

Pannok and Singhatanadgid (2006) studies the buckling behaviour of rectangular and skew thin composite plates with various boundary conditions using the Ritz method along with the proposed out-of-plane displacement functions. The boundary conditions considered in this study are combinations of simple support, clamped support and free edge. The out-of-plane displacement functions in form of trigonometric and hyperbolic functions are determined from the Kantorovich method.

Buket Okutan Baba (2007) studied the influence of boundary conditions on the buckling load for rectangular plates. Boundary conditions consisting of clamped, pinned, and their combinations are considered. Numerical and experimental studies are conducted to investigate the effect of boundary conditions, length/thickness ratio, and ply orientation on the buckling behaviour of E-glass/epoxy composite plates under in-plane compression load. Buckling analysis of the laminated composites is performed by using finite element analysis software ANSYS. Tests have been carried out on laminated composites with circular and semicircular cut-outs under various boundary conditions. Comparisons are made between the test results and predictions based on finite element analysis.

Pein and Zahari (2007) studied the structural behaviour of woven fabric composites subject to compressive load which is lacking. The main objective of this study is to carry out the experiment analysis for the 800gm woven glass-epoxy composite laminated plates with and without holes subjected to quasi-static compressive load. The ultimate load and the structural and material behaviour of the composite laminated plates under compression have also been studied. Finally, a parametric study is performed to investigate the effect of varying the fibre orientations and different central hole sizes onto the strength of the laminates.

A progressive failure analysis algorithm has been developed by . Zahari and Azmee (2008) and implemented as a user subroutine in a finite element code (ABAQUS) in order to model the non-linear material behaviour and to capture the complete compressive response of woven composite plates made of glass-epoxy material. Tsai-Hill failure theory has been employed in the progressive failure methodology to detect failure of the woven composite laminates.

Murat Yazici (2008) studied the influence of square cut-out upon the buckling stability of multilayered, steel woven fiber-reinforced polypropylene thermoplastic matrix composite plates are studied by using numerical and experimental methods. The laminated plates under uniform pressure are formed by stacking three composite layers bonded symmetrically. The FE and experimental results are presented for various fiber orientation angles and plate boundary conditions.

Aim and scope of study

Thus far, there have been numerous studies on the composite laminated structures which find widespread applications in many engineering fields namely aerospace, biomedical, civil, marine and mechanical engineering because of their ease of handling, good mechanical properties and low fabrication cost. They also possess excellent damage tolerance and impact resistance.

From the literature, it is evident that most of the studies are based on the numerical approach. Less attention has been paid on the buckling of composite plates. Due to the practical requirements, cutouts are often required in structural components due to functional requirements, to produce lighter and more efficient structures. Most stability studies of composite plates with cutout have focused on square plates under simply supported conditions to minimize the mathematical complexities.

From the literature review it was found that most of the studies were focused on unidirectional fibre. Industry driven woven fibers are being increasingly used in many industries. Hence we have to give more importance on its structural behaviour. It also indicate that the interaction among stacking sequence, cutout shape and length/thickness ratio on the buckling behaviour of woven fiber laminated composites are needed to investigate in more detail. The aim of performing this research is to extend the knowledge of the structural behaviour of woven fabric composites subject to compressive load which is lacking. The main objective of this study is to carry out the experiment analysis for the woven glass-epoxy composite laminated plates with and without holes subjected to static compressive load.

CHAPTER-2

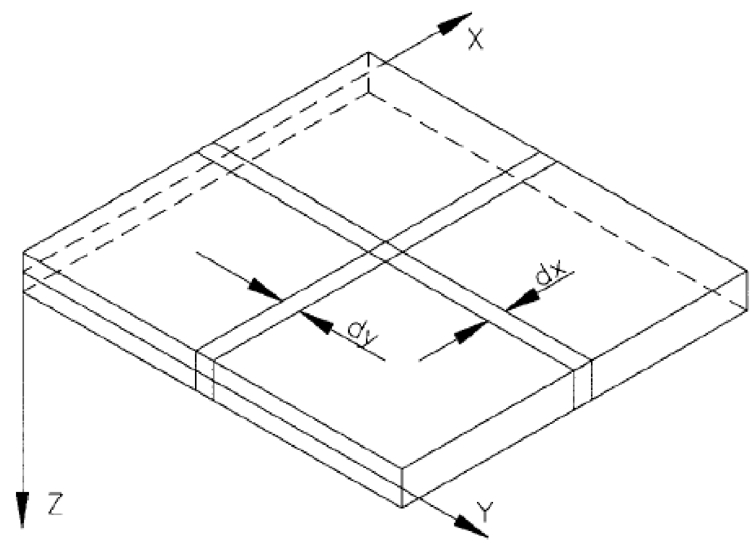
THEORETICAL FORMULATION

THEORETICAL FORMULATION

The buckling of a plate involves two planes, namely, xz,yz and two boundary conditions on each edge of the plate. The basic difference between plate and column lies in the buckling characteristics. The column, once it buckles, cannot resist any additional axial load. Thus, the critical load of the column is also its failure load. On the other hand, a plate, since it is invariably supported at the edges, continues to resist the additional axial load even after the primary buckling load is reached and does not fail even when the load reaches a value 10-15 times the buckling load.

Theory of bending of thin plates

The theory for thin plates is similar to the theory for beams. In pure bending of beams, "the stress distribution is obtained by assuming that cross-sections of the bar remain plane during bending and rotate only with respect to their neutral axes so as to be always normal to the deflection curve." For a thin plate, bending in two perpendicular directions occur. A rectangular plate element is shown below:



(fig.1.a) Thin plate notation

The basic assumptions of elastic plate bending are:

1. Perfectly flat plate and of uniform thickness.
2. The thickness of the plate is small compared with other dimensions. For plate bending, the thickness, t, is less than or equal to ¼ of the smallest width of the plate. For plate buckling equations, the thickness, *t*, should be 1/10 of the smallest width of the plate.
3. Deflections are small, i.e., smaller or equal to 1/2 of the thickness.
4. The middle plane of the plate does not elongate during bending and remains a neutral surface.
5. The lateral sides of the differential element, in the above figure, remain plane during bending and rotate only to be normal to the deflection surface. Therefore, the stresses and strains are proportional to their distance from the neutral surface.
6. The bending and twisting of the plate element resist the applied loads. The effect of shearing forces is neglected.

Buckling of composite plate

Composite materials consist of two or more materials which together produce desirable properties that cannot be achieved with any of the constituents alone. Fiber-reinforced composite materials, for example , contain high strength and high modulus fibers are the principal load carrying members, and the matrix material keeps the fibers together, act as a load-transfer medium between fibers from being exposed to the environment. The lay up sequence of unidirectionally reinforced “plies" as indicated in Fig.1. Each ply is typically a thin (approximately 0.2 mm) sheet of collimated fibers impregnated with an uncured epoxy or other thermosetting polymer matrix material. The orientation of each ply is arbitrary, and the layup sequence is tailored to achieve the properties desired of the laminate.

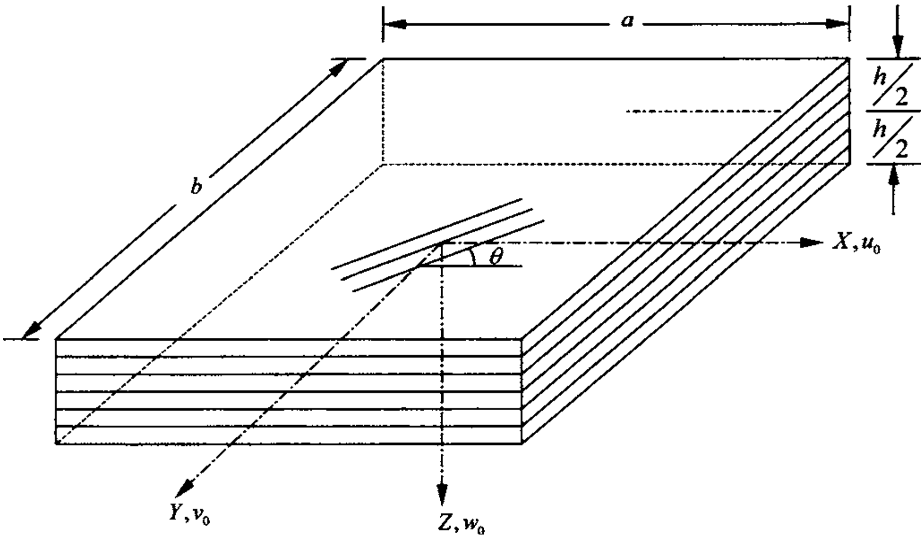
Fiber reinforced composite materials for structural applications are made in the form of a thin layer, called lamina. A lamina is a macro unit of material whose material properties are determined through appropriate laboratory tests. Structural elements such as bars, beams and plates are then formed by stacking the layers to achieve desired strength and stiffness. Fiber orientation in each lamina and stacking sequence of the layers can be choosen to achieve desired strength and stiffness .

Governing equation

A laminated plate is made by using a lamina as the building block. Its stiffness is obtained from the properties of the constituent laminae. To do this, we should know the orientations of the principle material directions of the laminae with respect to the laminate axis. Therefore , a knowledge of stress and strain through the laminate thickness is necessary.

We shall make the following assumptions regarding the behaviour of a laminate:

1. It is made up of perfectly bonded laminae.
2. The bonds are infinitesimally thin and no lamina can slip relative to the other. This implies that the displacements are continuous across the lamina boundaries. As a result, the laminate behave like a lamina with special properties.
3. After buckling, a line originally straight and perpendicular to the middle surface of the laminate remains straight and perpendicular to the middle surface.
4. The strain perpendicular to the middle surface of the laminate is ignored.



(Fig.1.b) Laminated composite plate under in-plane compression.

Classical Laminate Theory, has been used to derive the governing buckling equations for a plate subjected to in plane load. To derive the governing equations we have considered first the equilibrium of force and then the equilibrium of moment in a way as discussed below:

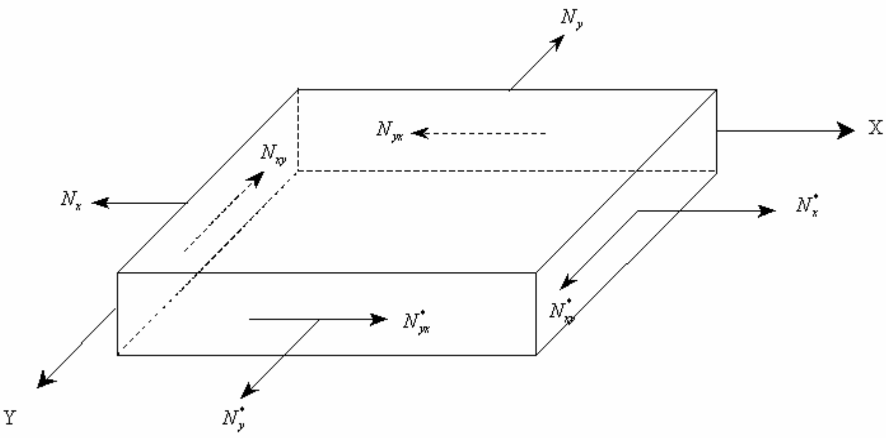
The equilibrium equations in terms of the forces (Fig. 2.a) are

$$\frac{\partial \mathrm{N}_x}{\partial x}+\frac{\partial N_{xy}}{\partial y}=0$$
$$\frac{\partial N_{xy}}{\partial x}+\frac{\partial N_y}{\partial y}=0 \qquad \text{............(1)}$$

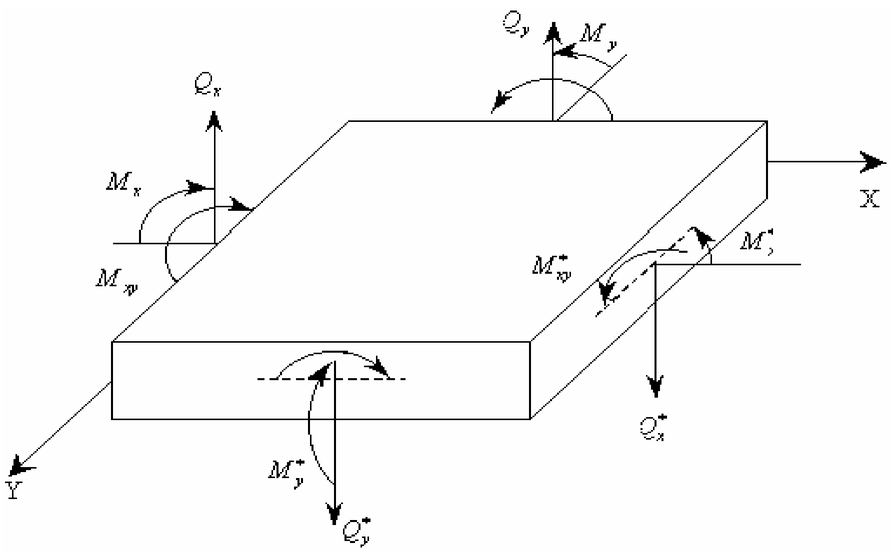
Where *Nx; Ny;* and *Nxy* are the internal forces in normal and tangential direction.
Again, the equilibrium equation in terms of the moments (Fig. 2.b) is

$$\frac{\partial^2 M_x}{\partial x^2}+2\frac{\partial^2 M_{xy}}{\partial x\partial y}+\frac{\partial^2 M_y}{\partial y^2}+N_x\frac{\partial^2 w}{\partial x^2}+N_y\frac{\partial^2 w}{\partial y^2}+2N_{xy}\frac{\partial^2 w}{\partial x\partial y}=0,\text{..........................(2)}$$

where, *Nx ;Ny; Nxy* are the forces applied at the edges.



(fig 2.a) Inplane forces on laminate



(fig.2.b) Moments on a laminate

The resultant forces *Nx; Ny* and *Nxy* and moments *Mx; My* and *Mxy* acting on a laminate are obtained by integration of the stress in each layer or lamina through the laminate thickness. Knowing the stress in terms of the displacement, we can obtain the stress resultants *Nx; Ny; Nxy; Mx ; My ;* and *Mxy*.

The stress resultants are defned as

$$N_x = \int_{\frac{-t}{2}}^{\frac{t}{2}} \sigma_x dz \qquad N_y = \int_{-\frac{t}{2}}^{\frac{t}{2}} \sigma_y dz \qquad N_{xy} = \int_{-\frac{t}{2}}^{\frac{t}{2}} \tau_{xy} dz$$

$$M_x = \int_{-\frac{t}{2}}^{\frac{t}{2}} \sigma_x zdz, \quad M_y = \int_{-\frac{t}{2}}^{\frac{t}{2}} \sigma_y zdz \quad M_{xy} = \int_{-\frac{t}{2}}^{\frac{t}{2}} \tau_{xy} zdz. \quad ....(3)$$

Where $\sigma_{x,}$ σ_y *and* τ_{xy} are normal and shear stress.

Actually, *Nx ;Ny* and *Nxy* are the force per unit length of the cross section of the laminate as shown in Fig.2.a. Similarly, *Mx ;My;* and *Mxy* are the moment per unit length as shown in Fig .2.b. Thus, the forces and moments for an *N*-layer laminate can be defined as

$$\begin{Bmatrix} N_x \\ N_y \\ N_{xy} \end{Bmatrix} = \int_{-\frac{h}{2}}^{\frac{h}{2}} \begin{Bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{Bmatrix}_r dz = \sum_{r=1}^{N} \int_{z_{r-1}}^{z_r} \begin{Bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{Bmatrix}_r dz, \qquad ....(4)$$

$$\begin{Bmatrix} M_x \\ M_y \\ M_{xy} \end{Bmatrix} = \int_{-\frac{h}{2}}^{\frac{h}{2}} \begin{Bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{Bmatrix}_r zdz = \sum_{r=1}^{N} \int_{z_{r-1}}^{z_r} \begin{Bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{Bmatrix}_r zdz, \qquad ....(5)$$

where, z_r and z_{r-1} are as defined in Fig. 3. Note that $z_0 = -\frac{t}{2}$ Substituting for σ_x, σ_y and τ_{xy} in equations (2) and (3) and integrating over the thickness of each layer and adding the results so obtained for *N* layers, we can write the stress resultants as

$$\begin{Bmatrix} N_x \\ N_y \\ N_{xy} \end{Bmatrix} = \begin{bmatrix} A_{11} & A_{12} & A_{16} \\ A_{12} & A_{22} & A_{26} \\ A_{16} & A_{26} & A_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_x^0 \\ \varepsilon_y^0 \\ \gamma_{xy}^0 \end{Bmatrix} + \begin{bmatrix} B_{11} & B_{12} & B_{16} \\ B_{12} & B_{22} & B_{26} \\ B_{16} & B_{26} & B_{66} \end{bmatrix} \begin{Bmatrix} k_x \\ k_y \\ k_{xy} \end{Bmatrix},$$

$$\begin{Bmatrix} M_x \\ M_y \\ M_{xy} \end{Bmatrix} = \begin{bmatrix} B_{11} & B_{12} & B_{16} \\ B_{12} & B_{22} & B_{26} \\ B_{16} & B_{26} & B_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_x^0 \\ \varepsilon_y^0 \\ \gamma_{xy}^0 \end{Bmatrix} + \begin{bmatrix} D_{11} & D_{12} & D_{16} \\ D_{12} & D_{22} & D_{26} \\ D_{16} & D_{26} & D_{66} \end{bmatrix} \begin{Bmatrix} k_x \\ k_y \\ k_{xy} \end{Bmatrix}, \qquad \text{..........(6) and(7)}$$

Where

$$A_{ij} = \sum_{r=1}^{N} \left(\overline{Q}_{ij}\right)_r (Z_r - Z_{r-1}),$$

$$B_{ij} = \frac{1}{2}\sum_{r=1}^{N} \left(\overline{Q}_{ij}\right)_r (Z_r^2 - Z_{r-1}^2), \qquad \text{..........(8)}$$

$$D_{ij} = \frac{1}{3}\sum_{r=1}^{n} \left(\overline{Q}_{ij}\right)_r \left(Z_r^3 - Z_{r-1}^3\right).$$

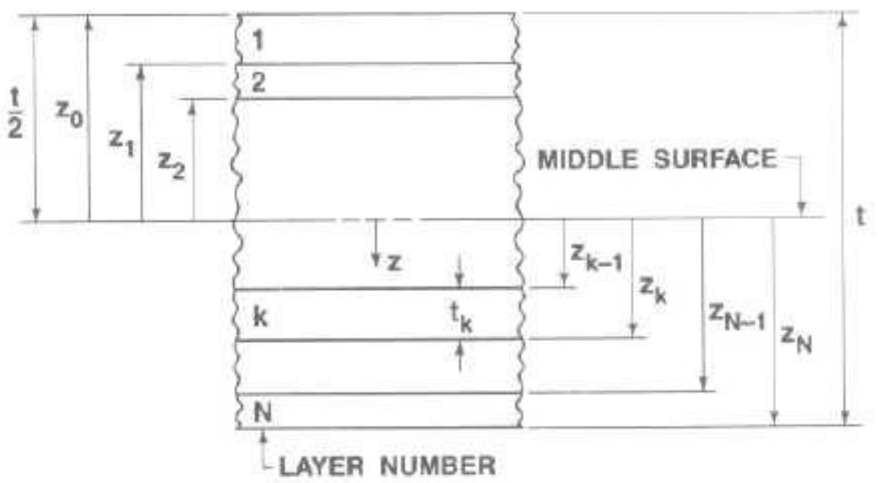


fig (3) Geometry of an N-layered laminate

Here, *Aij* are the extensional stiffness, *Bij* the coupling stiffness, and *Dij* the flexural stiffness. For antisymmetric angle-ply and cross-ply laminates stress resultants are simplified in the following sections:

In the case of angle-ply laminates where the fibre orientation θ alternates from lamina to lamina as $+\theta/-\theta/+\theta/-\theta$, the force and moment resultants are

$$\begin{Bmatrix} N_x \\ N_y \\ N_{xy} \end{Bmatrix} = \begin{bmatrix} A_{11} & A_{12} & 0 \\ A_{12} & A_{22} & 0 \\ 0 & 0 & A_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_x^0 \\ \varepsilon_y^0 \\ \gamma_{xy}^0 \end{Bmatrix} + \begin{bmatrix} B_{11} & B_{12} & B_{16} \\ B_{12} & B_{22} & B_{26} \\ B_{16} & B_{26} & B_{66} \end{bmatrix} \begin{Bmatrix} k_x \\ k_y \\ k_{xy} \end{Bmatrix},$$

$$\begin{Bmatrix} M_x \\ M_y \\ M_{xy} \end{Bmatrix} = \begin{bmatrix} B_{11} & B_{12} & B_{16} \\ B_{12} & B_{22} & B_{26} \\ B_{16} & B_{26} & B_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_x^0 \\ \varepsilon_y^0 \\ \gamma_{xy}^0 \end{Bmatrix} + \begin{bmatrix} D_{11} & D_{12} & D_{16} \\ D_{12} & D_{22} & D_{26} \\ D_{16} & D_{26} & D_{66} \end{bmatrix} \begin{Bmatrix} k_x \\ k_y \\ k_{xy} \end{Bmatrix}, \qquad \text{.......(9) and (10)}$$

Such a laminate is called an anti-symmetric angle-ply laminate. In this type of laminate, if each lamina has the same thickness, it is then called a regular anti-symmetric angle-ply laminate. For such a laminate, equations (6) and (7) reduce to

$$\begin{Bmatrix} N_x \\ N_y \\ N_{xy} \end{Bmatrix} = \begin{bmatrix} A_{11} & A_{12} & 0 \\ A_{12} & A_{22} & 0 \\ 0 & 0 & A_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_x^0 \\ \varepsilon_y^0 \\ \gamma_{xy}^0 \end{Bmatrix} + \begin{bmatrix} 0 & 0 & B_{16} \\ 0 & 0 & B_{26} \\ B_{16} & B_{26} & 0 \end{bmatrix} \begin{Bmatrix} k_x \\ k_y \\ k_{xy} \end{Bmatrix},$$

$$\begin{Bmatrix} M_x \\ M_y \\ M_{xy} \end{Bmatrix} = \begin{bmatrix} 0 & 0 & B_{16} \\ 0 & 0 & B_{26} \\ B_{16} & B_{26} & 0 \end{bmatrix} \begin{Bmatrix} \varepsilon_x^0 \\ \varepsilon_y^0 \\ \gamma_{xy}^0 \end{Bmatrix} + \begin{bmatrix} D_{11} & D_{12} & 0 \\ D_{12} & D_{22} & 0 \\ 0 & 0 & D_{66} \end{bmatrix} \begin{Bmatrix} k_x \\ k_y \\ k_{xy} \end{Bmatrix}, \qquad \text{...........(11) and (12)}$$

Let the laminae are oriented alternatively at 0^0 and 90^0. A laminate of this type is termed as a cross-ply laminate. Such a laminate can, again, be either symmetric cross-ply or anti-symmetric cross-ply.

Substituting for $N_x;N_y;N_{xy};M_x;M_y;M_{xy}$ from equations (9) and (10), after substituting for $\varepsilon_x^0, \varepsilon_y^0 and \gamma_{xy}^0$ $k_x; k_y; k_{xy}$ in equations (1) and (2), we get the governing equations as

$$A_{11}\frac{\partial^2 u^0}{\partial x^2}+(A_{12}+A_{66})\frac{\partial^2 v^0}{\partial x\partial y}+A_{16}(\frac{\partial^2 u^0}{\partial x^2}+2\frac{\partial^2 u^0}{\partial x\partial y})+A_{26}\frac{\partial^2 v^0}{\partial y^2}+A_{66}\frac{\partial^2 u^0}{\partial y^2}-B_{11}\frac{\partial^3 w}{\partial x^3}-3B_{16}\frac{\partial^3 w}{\partial x^2\partial y}$$
$$-(B_{12}+2B_{66})\frac{\partial^3 w}{\partial x\partial y^2}-B_{26}\frac{\partial^3 w}{\partial y^3}=0$$

$$A_{16}\frac{\partial^2 u^0}{\partial x^2}+(A_{12}+A_{66})\frac{\partial^2 u^0}{\partial x\partial y}+A_{26}\frac{\partial^2 u^0}{\partial y^2}+A_{66}\frac{\partial^2 v^0}{\partial x^2}+2A_{26}\frac{\partial^2 v^0}{\partial x\partial y}+A_{22}\frac{\partial^2 v^0}{\partial y^2}-B_{16}\frac{\partial^3 w}{\partial x^3}$$
$$-(B_{12}+2B_{66})\frac{\partial^3 w}{\partial x^2\partial y}-3B_{26}\frac{\partial^3 w}{\partial x\partial y^2}-B_{22}\frac{\partial^3 w}{\partial y^3}=0$$

$$D_{11}\frac{\partial^4 w}{\partial x^4}+4D_{16}\frac{\partial^4 w}{\partial x^3\partial y}+(2D_{12}+4D_{66})\frac{\partial^4 w}{\partial x^2\partial y^2}+4D_{26}\frac{\partial^4 w}{\partial x\partial y^3}+D_{22}\frac{\partial^4 w}{\partial y^4}-B_{11}\frac{\partial^3 u^0}{\partial x^3}$$
$$-3B_{16}\frac{\partial^3 u^0}{\partial x^2\partial y}-(B_{12}+2B_{66})\frac{\partial^3 u^0}{\partial x\partial y^2}-B_{26}\frac{\partial^3 u^0}{\partial y^3}-B_{16}\frac{\partial^3 v^0}{\partial x^3}-(B_{12}+2B_{66})\frac{\partial^3 v^0}{\partial x\partial y^2}$$
$$-(B_{12}+2B_{66})\frac{\partial^3 v^0}{\partial x\partial y^2}-B_{22}\frac{\partial^3 v^0}{\partial y^3}=-N_x\frac{\partial^2 w}{\partial x^2}-N_y\frac{\partial^2 w}{\partial y^2}-2N_{xy}\frac{\partial^2 w}{\partial x\partial y}$$

For a general laminate, all the above three equations, have to be solved simultaneously as they are coupled.

CHAPTER-3

EXPERIMENTAL STUDY

EXPERIMENTAL STUDY

In view of difficulty of theoretical and numerical analysis for laminated structure behaviours, experimental methods have become important in solving the buckling problem of laminated composite plates. The experimental and numerical analysis done on aluminium plate showed an appreciable match in the results. Taking the above proof for the correctness of the experimental procedure. Here the same experimental procedure was followed for a composite plate.

To understand the effect of cut out shape, length/thickness ratio, ply orientation, and length/breadth ratio on the compressive behaviour of woven glass epoxy laminated composite plates compression test was performed. The specimen was clamped at two side and kept free at other two sides The specimens were loaded in axial compression by using a tensile testing machine of 100 tonne load capacity.

The buckling load is determined from the load –displacement curve. In this study both out of plane displacement and end shortening of the plate was plotted against the applied load. Buckling load is determined from the intersection point of two tangent drawn from the pre buckling and post buckling regions. In this study buckling load of composite plate determined by using the above two method and compared with each other.

Buckling Experiment of aluminium plate

In this study buckling load of aluminium plates determined numerically and experimentally. Two different plate length were used: 300mm and 200mm. The width and thickness of the plates are 200mm and 1.7mm respectively. The Youngs modulus value was 70000N/mm^2 and poissons ratio was taken as 0.3.

Test procedure

The specimen were loaded in axial compression using a Instron tensile testing machine of 100 KN capacity. The specimen was clamped at two ends and kept free at the other two ends. A dial gauge was mounted at the centre of the specimen to observe the lateral deflection. All specimen were loaded slowly until buckling. The experimental set up is shown below. Clamped boundary conditions were simulated along the top and bottom edges, restraining 40mm length .For axial loading, the test specimens were placed between the two extremely stiff machine heads, of which the lower one was fixed during the test, whereas the upper head was moved downwards by servo hydraulic cylinder. All plates were loaded at constant cross-head speed of 1mm/min. The test set up is shown below.

fig.(4.a) Instron testing machine

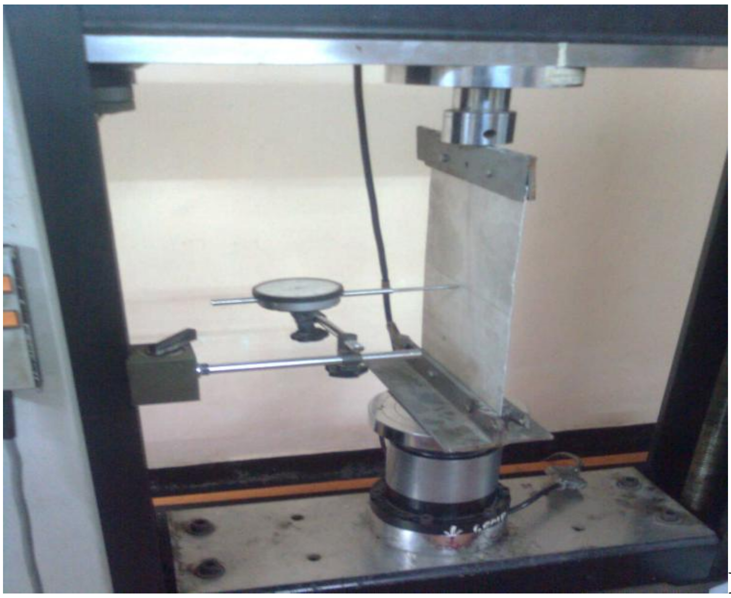


Fig. (4.b)

As the load was increased the dial gauge needle started moving, and at the onset of buckling there was a sudden large movement of the needle. The load corresponding to this point will be the buckling load of the specimen. The load v/s displacement curve and load v/s end shortening curve was plotted. The displacement is plotted on the x -axis and load was plotted on the y- axis. The load, which is the initial part of the curve deviated linearity, is taken as the critical buckling load. That point is determined from the intersection of two tangent drawn from the pre-buckling and post-buckling regions.

Composite Specimen Preparation and Manufacturing

To meet the wide range of needs which may be required in fabricating composites, the industry has evolved oven a dozen separate manufacturing processes as well as a number of hybrid processes. Each of these processes offers advantages and specific benefits which may apply to the fabricating of composites. Hand lay-up and spray-up are two basic moulding processes. The hand lay-up process is the oldest, simplest, and most labour intense fabrication method. The process is most common in FRP marine construction. In hand lay-up method liquid resin is placed along with reinforcement (woven glass fiber) against finished surface of an open mould. Chemical reactions in the resin harden the material to a strong, light weight product. The resin serves as the matrix for the reinforcing glass fibers, much as concrete acts as the matrix for steel reinforcing rods. The percentage of fiber and matrix was 50:50 in weight.

(Fig.5.a) Glass fiber

(fig. 5.b) Plate casting

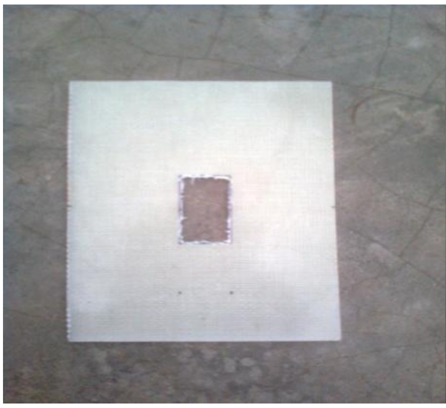
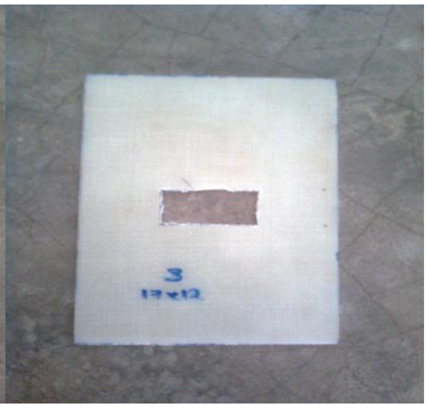
Contact moulding in an open mould by hand lay-up was used to combine plies of WR in the prescribed sequence. A flat plywood rigid platform was selected. A plastic sheet was kept on the plywood platform and a thin film of polyvinyl alcohol was applied as a releasing agent by use of spray gun. Laminating starts with the application of a gel coat (epoxy and hardener) deposited on the mould by brush, whose main purpose was to provide a smooth external surface and to protect the fibers from direct exposure to the environment. Ply was cut from roll of woven roving. Layers of reinforcement were placed on the mould at top of the gel coat and gel coat was applied again by brush. Any air which may be entrapped was removed using serrated steel rollers. The process of hand lay-up was the continuation of the above process before the gel coat had fully hardened. Again, a plastic sheet was covered the top of plate by applying polyvinyl alcohol inside the sheet as releasing agent. Then, a heavy flat metal rigid platform was kept top of the plate for compressing purpose. The plates were left for a minimum of 48 hours before being transported and cut to exact shape for testing. The following constituent materials were used for fabricating the plate:

1. E-glass woven roving as reinforcement
2. Epoxy as resin
3. Hardener as catalyst
4. Polyvinyl alcohol as a releasing agent

After 48 hours curing the specimen were cut in to desired sizes ,with and without cutout shown in fig. Circular, square, and rectangular cutout of same area(9.62cm^2)were made for the experiments. The specifications of plate tested in the present study shown in the table-1 and the plate with various cut out shape was shown in fig.6. The mechanical properties of the composite plates were determined by Instron tensile testing machine of 100KN load capacity. A specimens whose fiber direction coincide with the loading direction was used to obtain the modulus of elasticity along the fiber direction. The specimen were loaded step-by-step up to rupture by the test machine. Strains in the fiber (ε_1) and transverse directions ((ε_2) were measured,by using these strains E values are determined.

The mechanical properties of the tested specimen obtained as E_{11}=7.7Gpa, E_{22}=7.7Gpa, $\nu_{12} = 0.12$ and G_{12} = 2.81Gpa.


circular square

Rectangular Rectangular

(Fig.6) The plate with different cutout shape tested in the present study

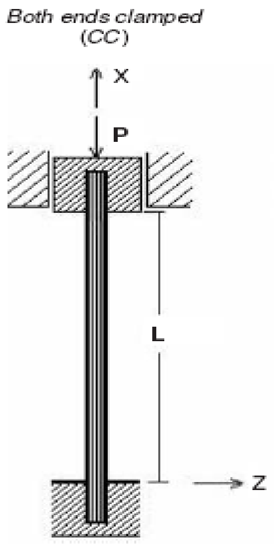
Plate no:	Stacking sequence	Length(mm)	Thickness (mm)	Width(mm)
Plate1	$[0]_4$	130	1.5	120
	$[0]_4$	130	1.32	120
	$[0]_4$	130	1.33	120
Plate2	$[0]_6$	130	2.25	120
	$[0]_6$	130	2.23	120
	$[0]_6$	130	2.21	120
Plate3	$[0]_8$	130	2.96	120
	$[0]_8$	130	3.37	120
	$[0]_8$	130	3.31	120
Plate4	$[0]_8$	175	3.31	120
	$[0]_8$	175	3.2	120
	$[0]_8$	175	3.15	120
Plate5	$[0]_8$	200	3.2	120
	$[0]_8$	200	3.1	120
Plate6	$[30/-30]_8$	130	3.31	120
	$[30/-30]_8$	130	3.28	120
	$[30/-30]_8$	130	3.25	120
Plate7	$[45/-45]_8$	130	3.24	120
	$[45/-45]_8$	130	3.24	120
Plate8	$[0]_8$	60	3.3	120
	$[0]_8$	60	3.28	120
Plate9	$[0]_{10}$	130	3.65	120
	$[0]_{10}$	130	3.65	120
Plate10	$[0]_{12}$	130	3.85	120
	$[0]_{12}$	130	3.83	120

Table 1 : Plate tested in the present study.

Buckling Experiments for composite plates

The specimen were loaded in axial compression using a uniaxial tensile testing machine of 100 tonne capacity. It is shown in fig.8.a. The specimen was clamped at two ends and kept free at the other two ends. A dial gauge was mounted at the centre of the specimen to observe the lateral deflection. All specimens were loaded slowly up to failure. Clamped boundary conditions were simulated along the top and bottom edges, restraining 40mm length .For axial loading, the test specimens were placed between the two extremely stiff machine heads, of which the lower one was fixed during the test, whereas the upper head was moved downwards by servo hydraulic cylinder. The shape of the plate after buckling was shown in fig.8.b. All laminated plates were loaded at constant cross-head speed of 200Kn/min. Finally the microscopic scanning was performed for the failed samples by using the JEOL-JSM-6480LV Scanning Electron Microscopic device shown in fig.9. The image of failed specimen obtained after scanning was shown in fig.10.

As the load was increased the dial gauge needle started moving, and at the onset of buckling there was a sudden large movement of the needle. The load corresponding to this point will be the buckling load of the specimen. The load v/s displacement curve and load v/s end shortening curve was plotted. The displacement is plotted on the x -axis and load was plotted on the y- axis. The load, which is the initial part of the curve deviated linearity,is taken as the critical buckling load. That point is determined from the intersection of two tangent drawn from the pre-buckling and post-buckling regions.

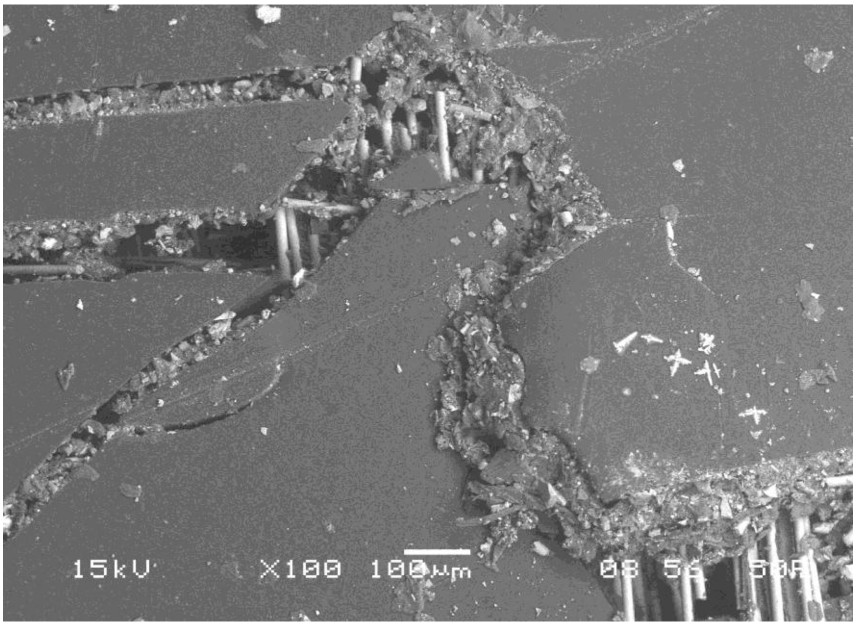


(fig.7) . Test set up for both side clamped boundary condition

(Fig 7.a)Before Buckling

(fig.7.b) After Buckling

Scanning Electron Microscope (fig.8)



Microscopoic image of the failure (fig.9)

Numerical Analysis

ANSYS was used to carry out the finite element analysis in the work. ANSYS is used to analyse the critical buckling load aluminium plates of different sizes. The dimension of the specimen were 300*200* 1.7mm and 200*200*1.7mm in length, width and thickness .Eigen value buckling analysis in ANSYS has four steps:

1.Build the model : It includes defining element type, real constants, material properties and modelling. In this study shell ,Elastic 8node 93 selected as the element type.

2.Solution(Static Analysis): It includes applying boundary conditions, applying loads and solving the static analysis. The applied boundary condition and load is shown below.

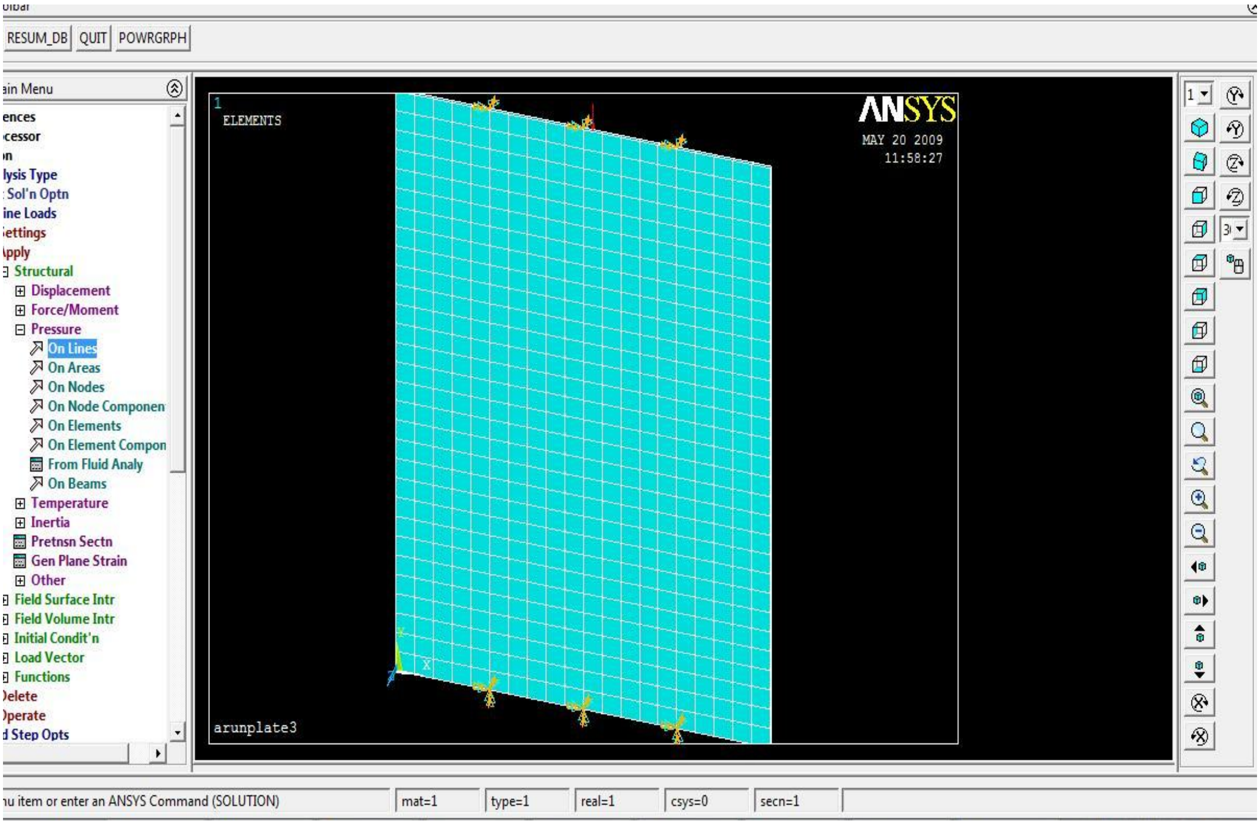


Fig.10

3. Eigen buckling analysis: Eigenvalue buckling analysis predicts the theoretical buckling strength of an ideal linear elastic structure.

4. Postprocessor: This steps includes listing buckling loads and viewing buckled shapes. We can plot the deformed and undeformed shape of the plate.

CHAPTER- 4

RESULTS AND DISCUSSION

RESULTS AND DISCUSSION

The buckling load for clamped- free aluminium plate determined. The results were both experimental analysis and finite element analysis. The agreement between the two method was generally good. The critical buckling loads obtained experimentally and by ANSYS is shown in the table2 .

plate No.	Length mm	Width mm	Thickness mm	Experimental Buckling load(N/mm)	ANSYS Buckling load(N/mm)
Plate-1	300	200	1.7	11.75	13.44
Plate-2	200	200	1.7	24.25	30.55

Table 2 : Buckling Results of Aluminium Plate

It was observed that the buckling load of plate-1 obtained from experimental and numerical analysis are identical and less than that of plate-2.The experimental buckling loads for both specimens are less than the ANSYS results. The deformed and undeformed shape of the aluminium plate was shown in the fig.12.

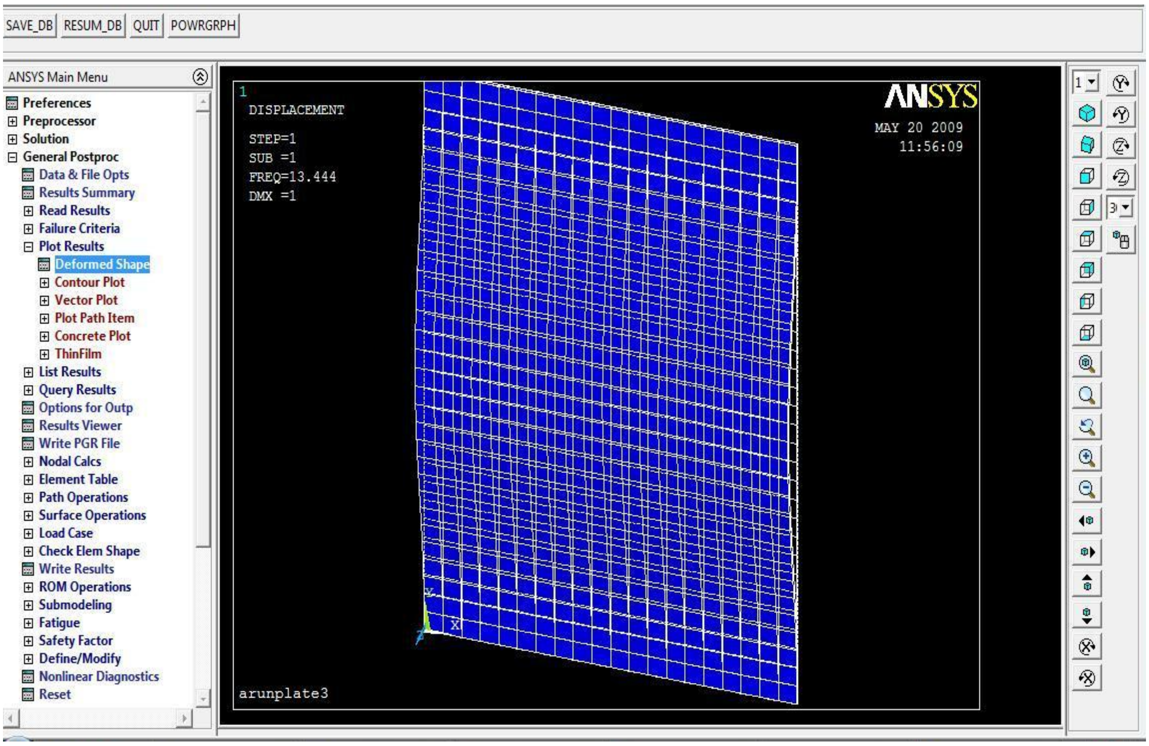


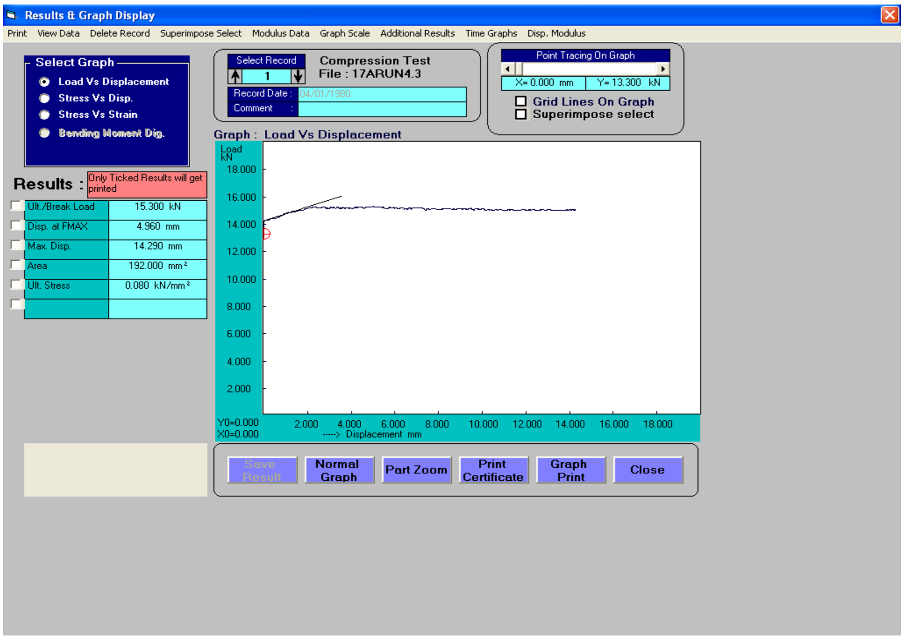
Fig.11

It is observed that all the laminated plates buckled globally until complete failure occurred as expected. Experimental buckling loads were identified from the load-displacement plots, and load- end shortening graph. Following graphs shows the load versus end-shortening curves and load versus out of plane displacement for the tested panels with and without central holes, different angle orientations, different aspect ratio and different length/thickness ratio. It is interesting to note that all the tested panels behave in a similar fashion where, their behaviour is almost linear initially before reaching the peak loads. Beyond those peak points of the load-displacement curves, majority of the laminates experienced large displacements before failure.

From the electron microscopic scanning it was found that, the bonding between the matrix and glass laminate was breaked during the failure. The glass fiber was came out from the plate.

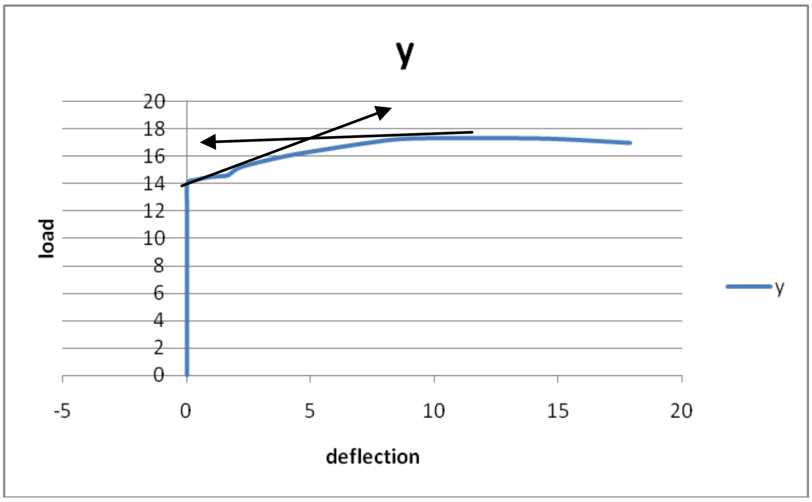
The load v/s out-of-plane displacement (4layer,130*120*1.32mm, $[0/0/0/0]_4$) and load v/s end shortening for Plate-1 are shown in fig.12 & 13 respectively. From the graph we can find that the displacement of the plate started at a load of 14KN, after which the plate was deflecting outward with an increase in the load. The thickness of the plate is very small. Hence the plate shows a large deflection for small increment in the load. The critical buckling load obtained from both the graphs was almost equal.

(Fig.12) Load versus out of plane displacement graph

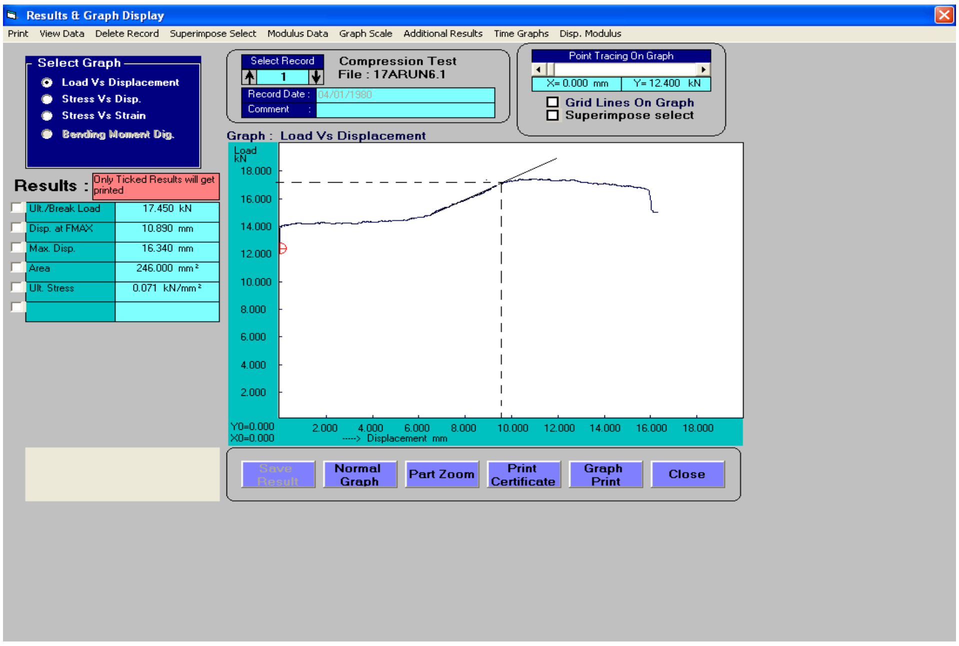


(Fig.13) Load versus end shortening graph

The load v/s out of plane displacement graph for Plate-2 (6 layer, 130*120*2.23mm.$[0/0]_6$) is shown in fig.14. The critical buckling load was determined from the intersection of tangents drawn from the pre-buckled region and post buckled region. The load v/s end shortening graph obtained from the test is shown in fig.15.The buckling load obtained from both graphs was almost equal.

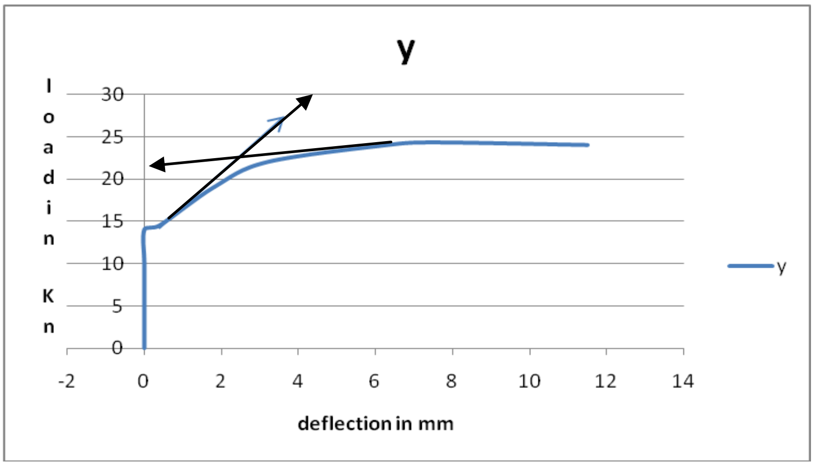


(Fig.14) Load-out of plane displacement

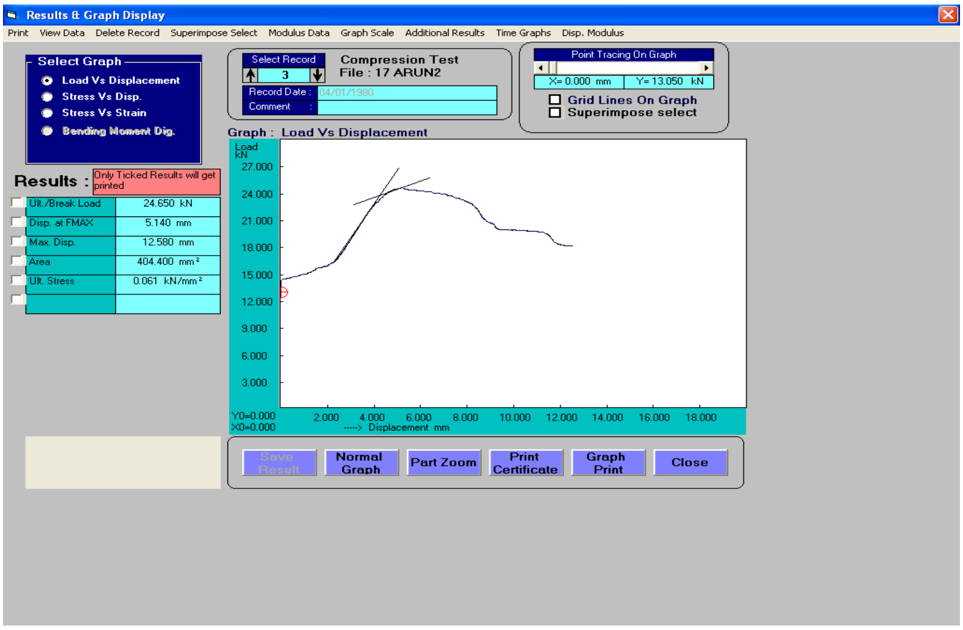


(Fig.15) Load- end shortening

The load v/s out of plane displacement graph for Plate-3 (8 layer ,130*120*3.31mm $[0/0/0/0]_8$) is shown in fig.16. The displacement started at 15KN load and the buckling occurred between 20 and 25KN. The load v/s end shortening graph is shown in fig. 17. The buckling load obtained from the load v/s end shortening plot was slightly different from load v/s out of plane displacement.

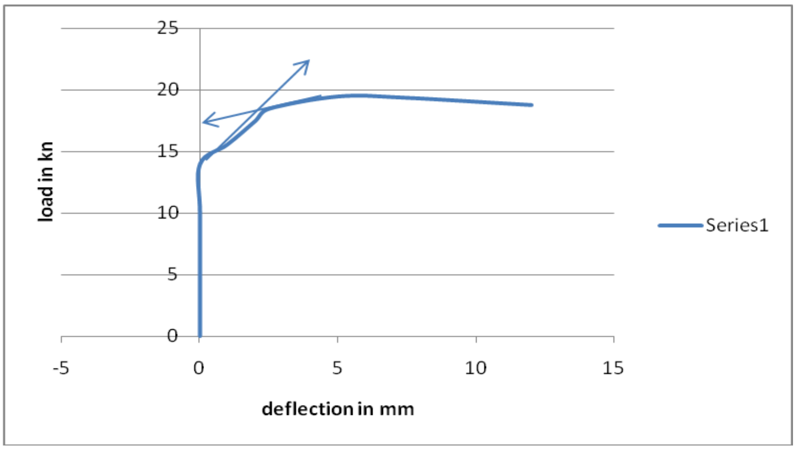


(Fig.16) Load v/s out of plane displacement

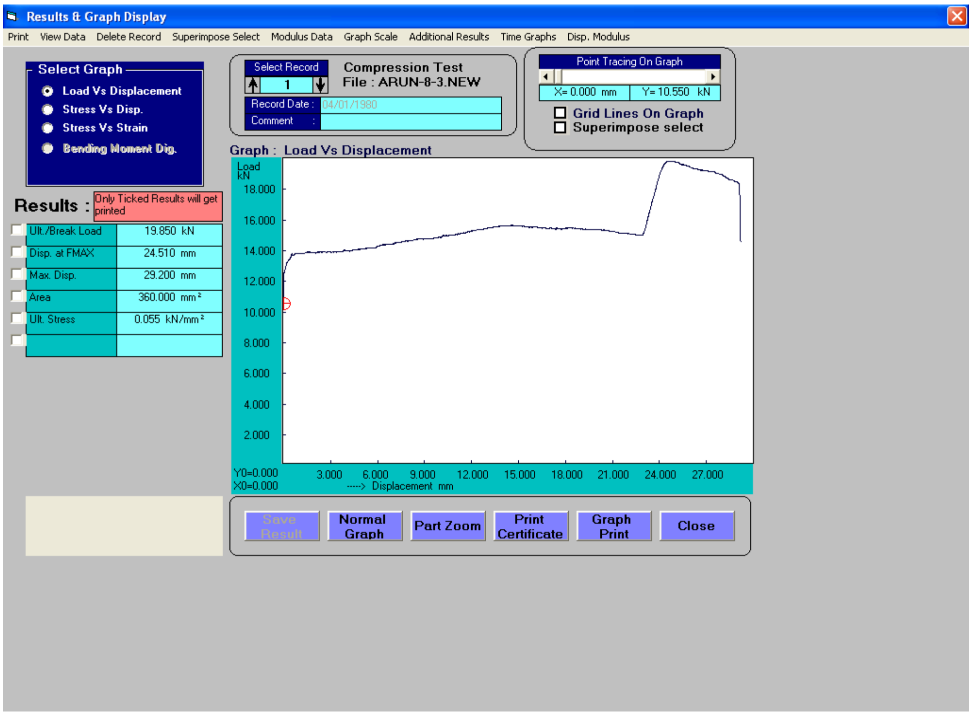


(fig.17) Load v/s end shortening

In Plate-4(8 layer, 175*120*3.31mm, $[0/0]_8$), the aspect ratio was changed by varying the length. The plot of load v/s out of plane displacement and load v/s end shortening is shown in fig. 18 & 19 respectively.

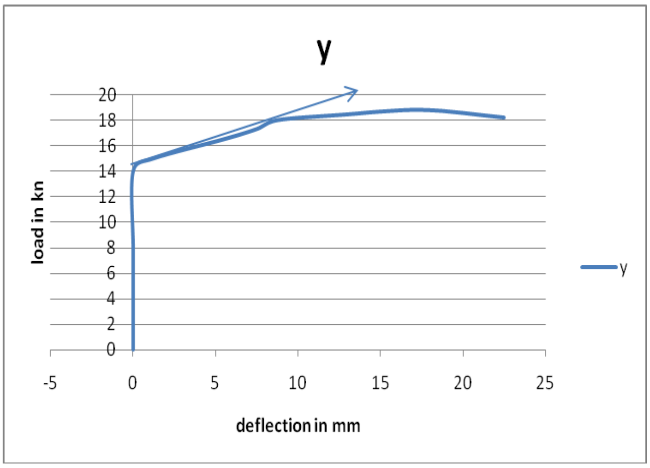


(fig.18) Load v/s out of plane displacement

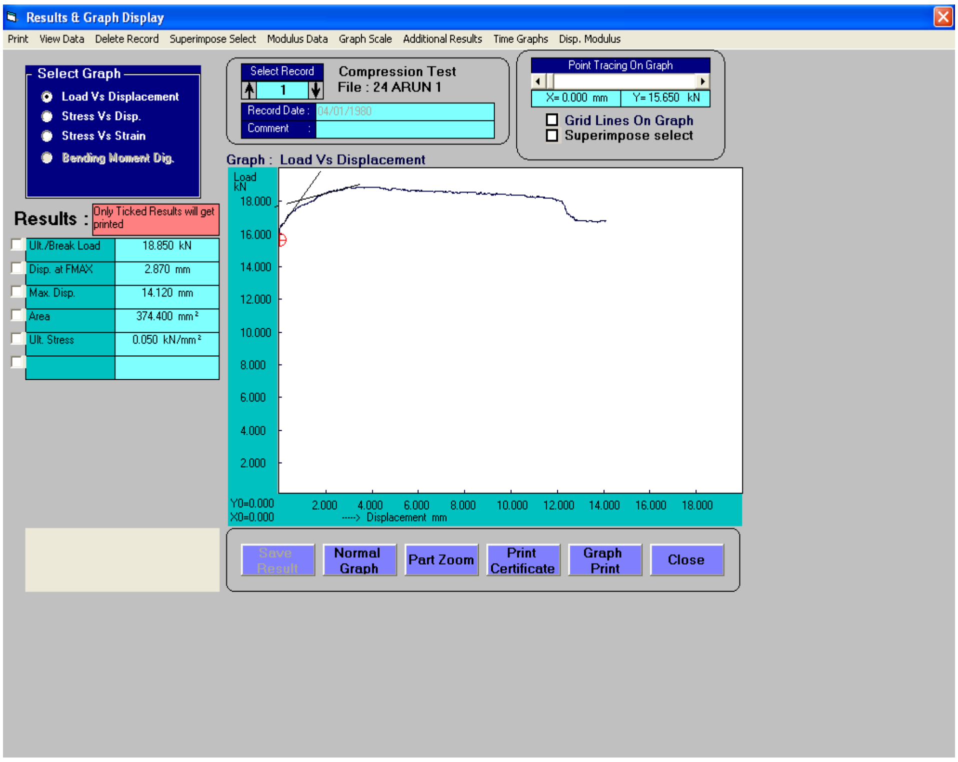


(fig.19) Load-end shortening

In Plate-5(8layer, 200*120*3.2mm, $[0/0/0/0]_4$),the length was again increased to study the effect of aspect ratio on buckling load. The load v/s out of plane displacement and load v/s end shortening plotted for plate 5 is shown in fig.20 & 21 respectively.

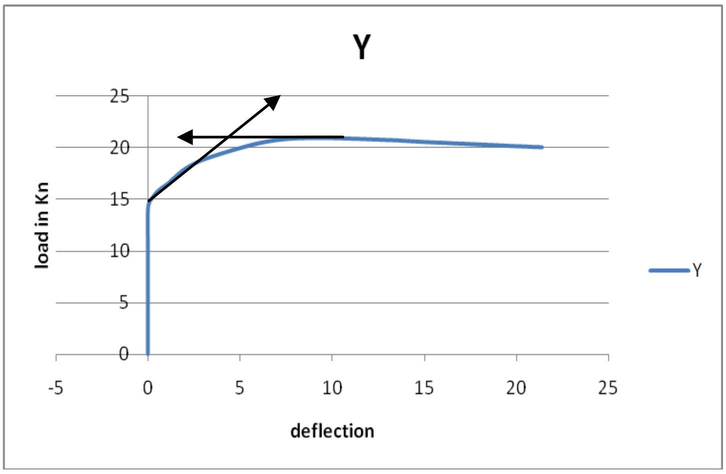


(Fig.20) Load –out of plane displacement

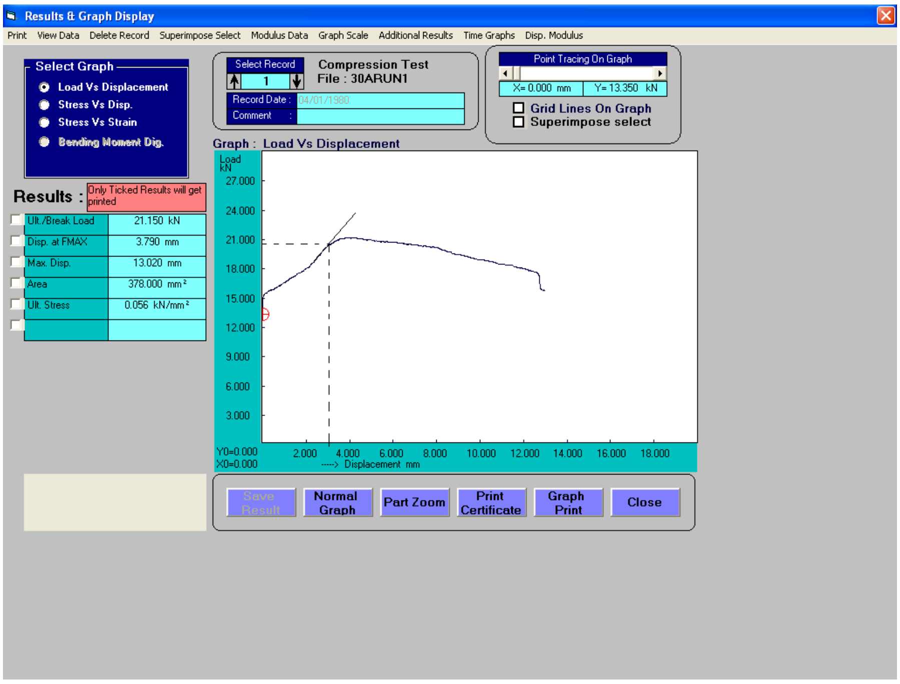


(fig.21) Load v/s end shortening

In Plate-6 (8 layer,130*120*3.31 mm, $[30/-30/30/-30]_8$) , the fiber orientation was changed from 0^0 to 30^0 .The buckling load was determined from fig.22 & fig.23 . It was found to be less compared to buckling load of plate with 0^0 orientation.

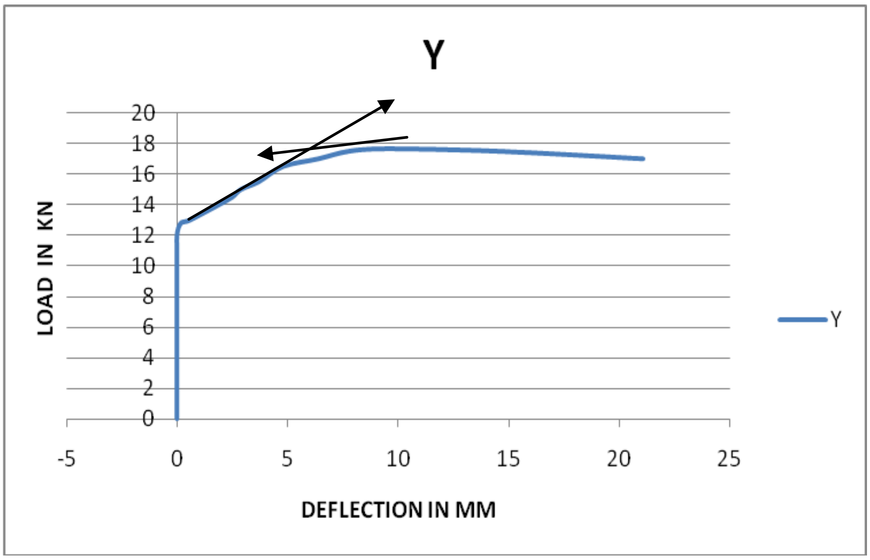


(Fig.22) Load v/s out of plane displacement

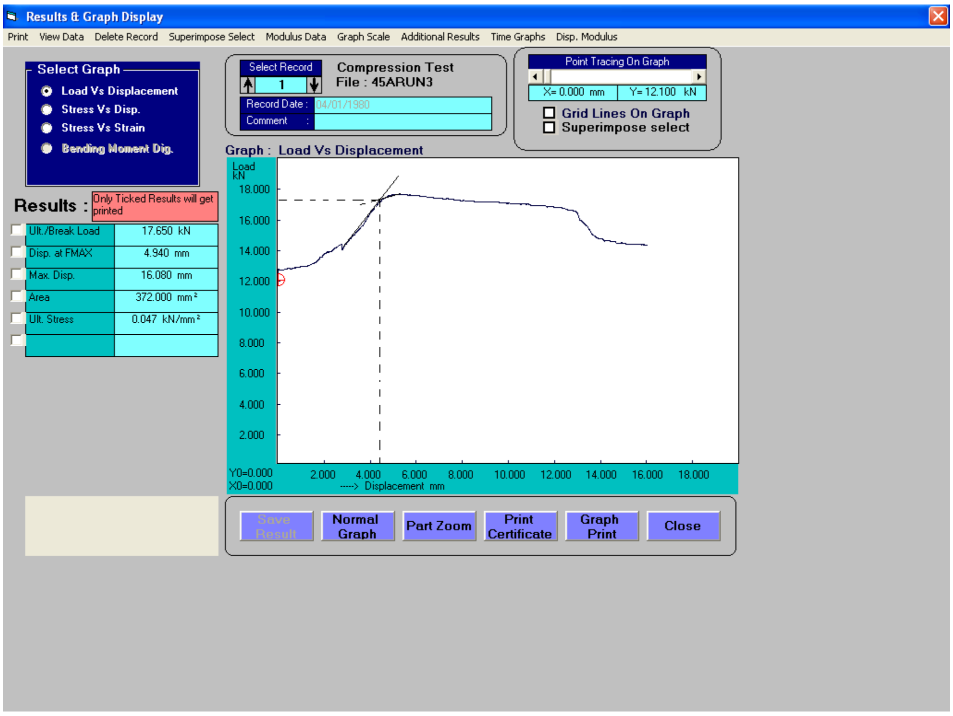


(fig.23) Load v/s end shortening

In Plate:-7(8layer, 130*120*3.24mm, $[45/-45/45/-45]_8$) the orientation was again changed to 45^0. The corresponding variation in load displacement graph is shown below.

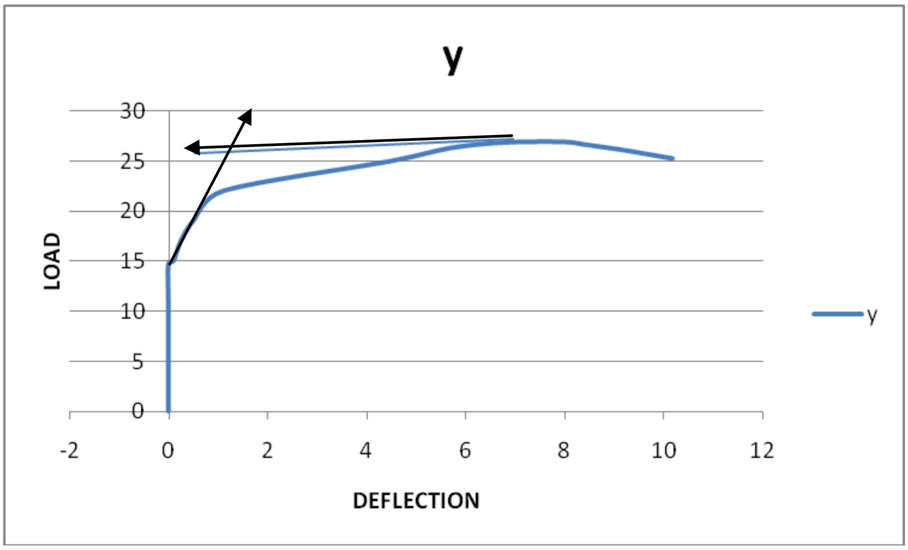


(fig.23) Load-out of plane displacement

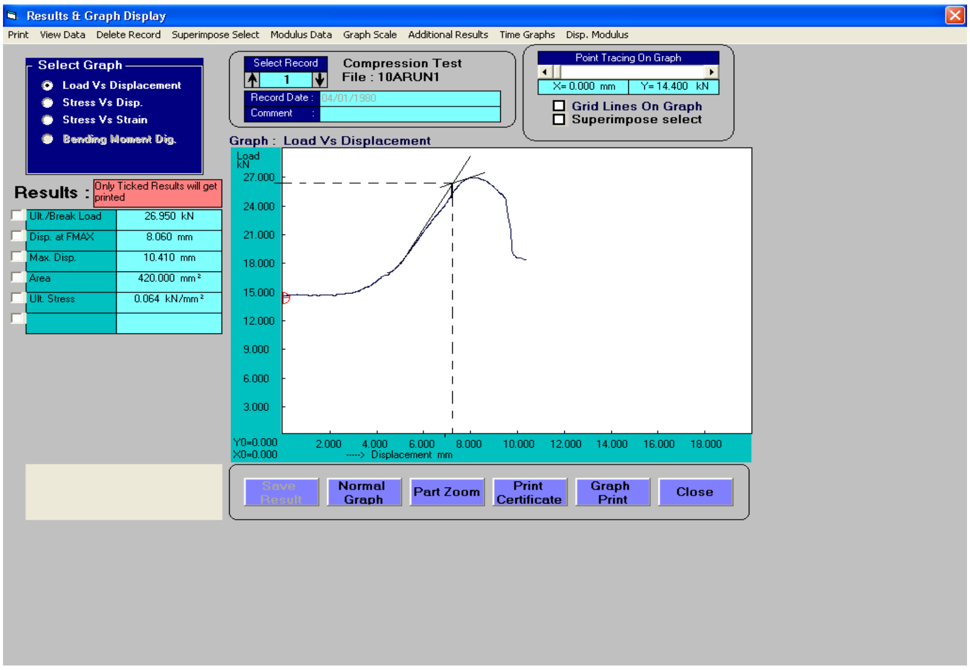


(fig.24) Load-end shortening

In Plate-8 (10 layer, 130*120*3.5mm, $[0/0/0/0]_{10}$) the number of layers was increased to 10. Hence the thickness of the plate increased. The load v/s out of plane displacement and load v/s end shortening is shown below. The buckling load obtained from both graphs was nearly equal.

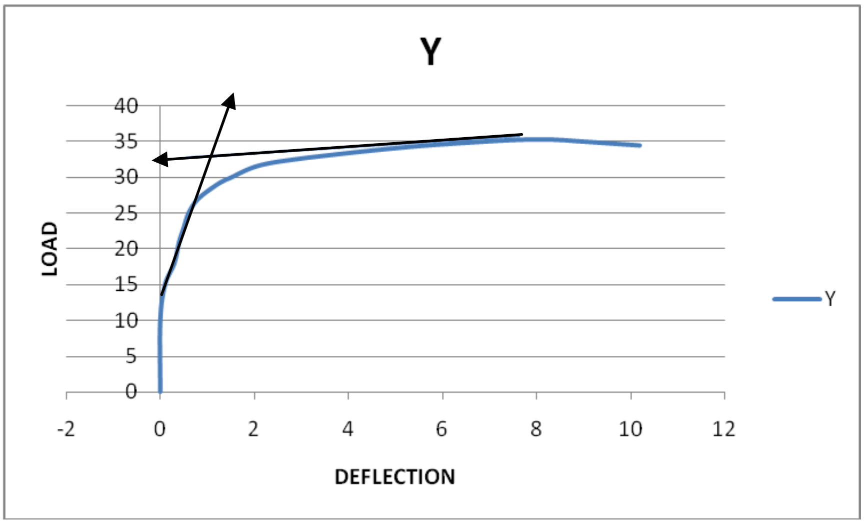


Load-out of plane displacement graph

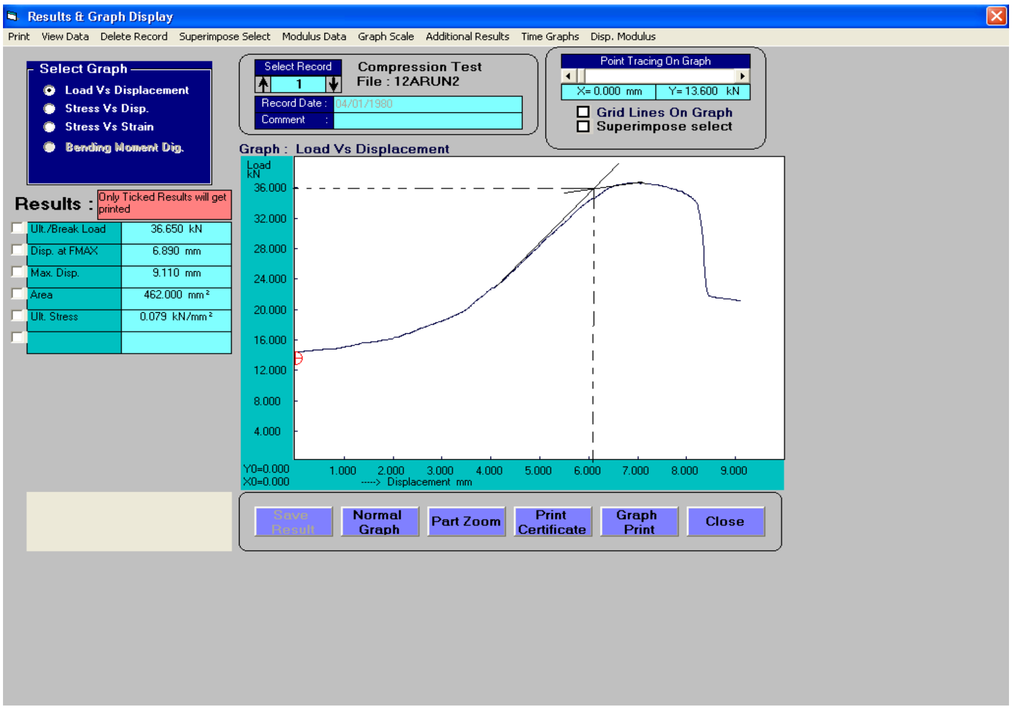


Load v/s end shortening graph

In Plate-9 (12 layer ,130*120* 3.8mm, $[0/0/0/0]_{12}$) the thickness was again increased by increasing the number of layers. Here 12 layers are used. The plots for load v/s out of plane displacement and load v/s end shortening is shown in fig.27 and 28 respectively.

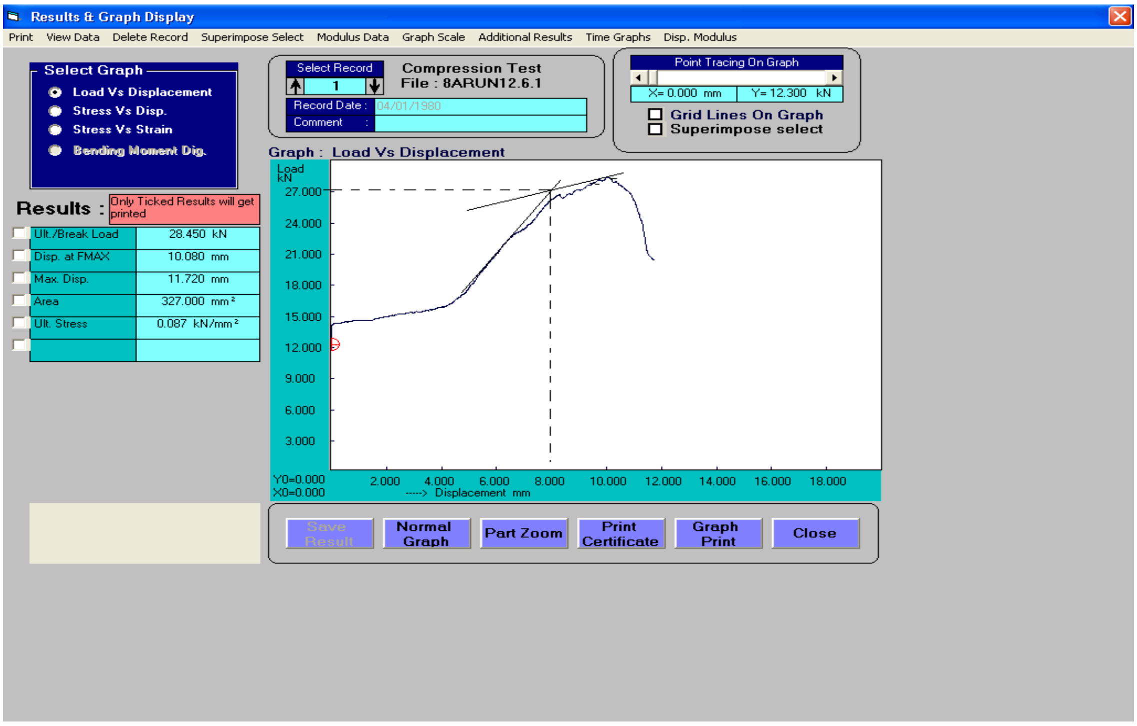


(Fig.27) Load- out of plane displacement



(fig.28) Load –end shortening graph

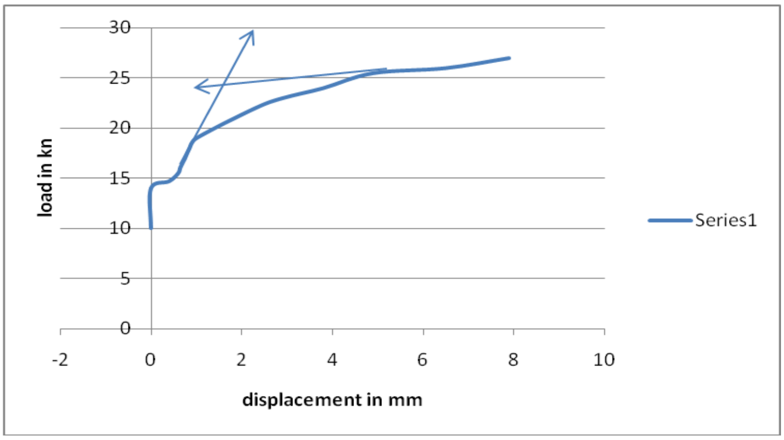
For Plate-10 (8 layer,60*120*3.1mm, $[0/0/0/0]_8$), it was difficult to fit the dial gauge with the plate due to its short length. Hence only load v/s end shortening was plotted and is shown in fig.29 .



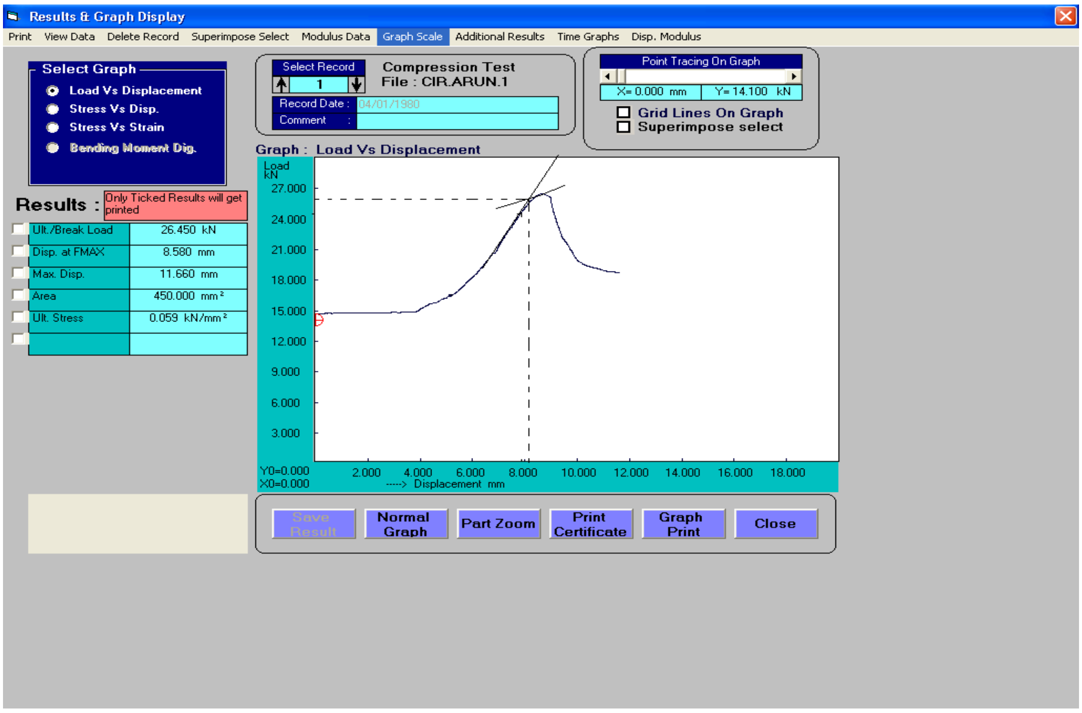
(Fig.29) Load-end shortening

Plate with circular cut out

In plate-11 (12 layer,130* 120*3.75mm, $[0/0/0/0]_{12}$) a circular cutout was made at the centre. It results in decrease in the buckling load. The load v/s out of plane displacement and load v/s end shortening is shown in the fig.30 and 31respectively.

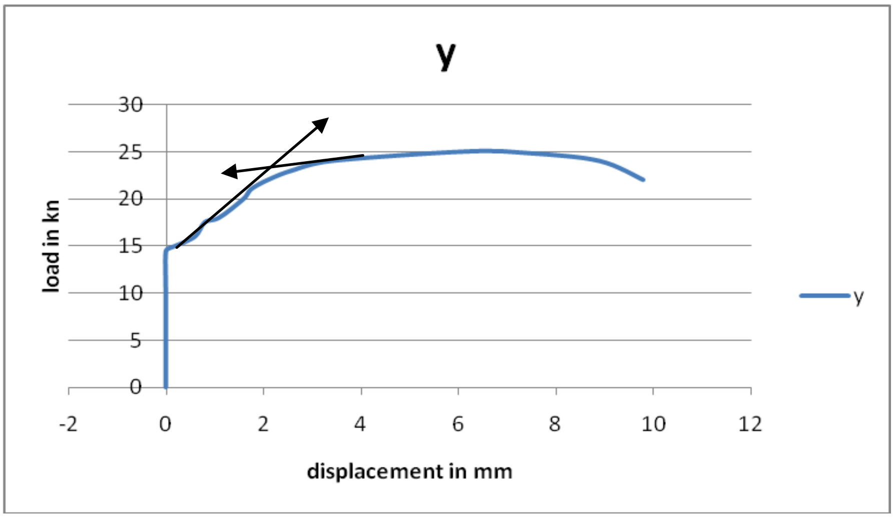


(Fig.30) Load v/s out of plane displacement

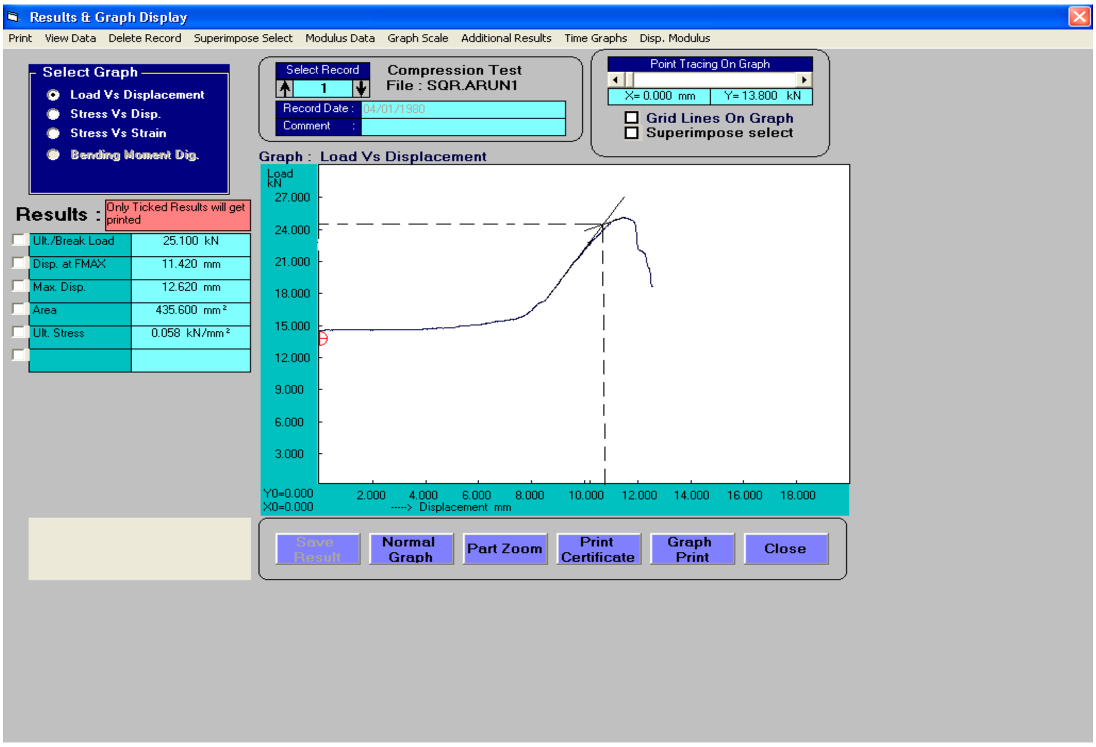


(fig.31) Load v/s end shortening

Here the load v/s displacement graph for plate with square cut out is given. The buckling determined from load v/s out of plane displacement and load v/s end shortening for Plate-12 (12 layer,130*120*3.65mm, $[0/0/0/0]_{12}$ **)** was almost equal .

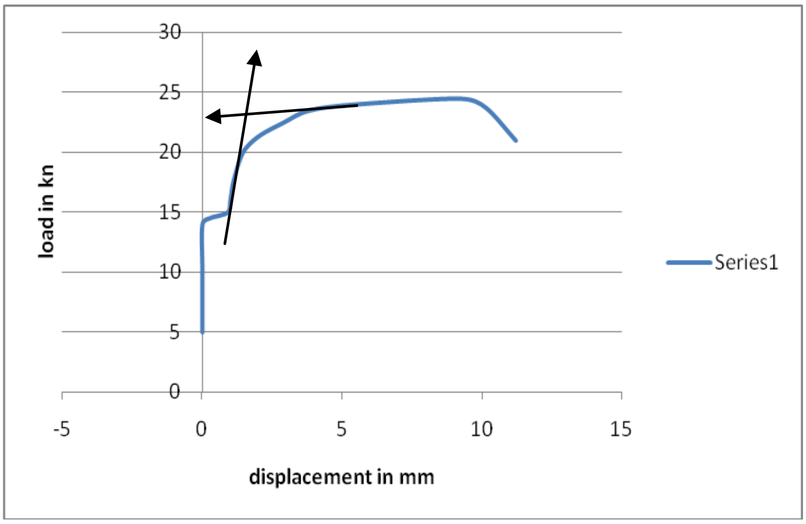


(Fig.32) Load v/s out of plane displacement

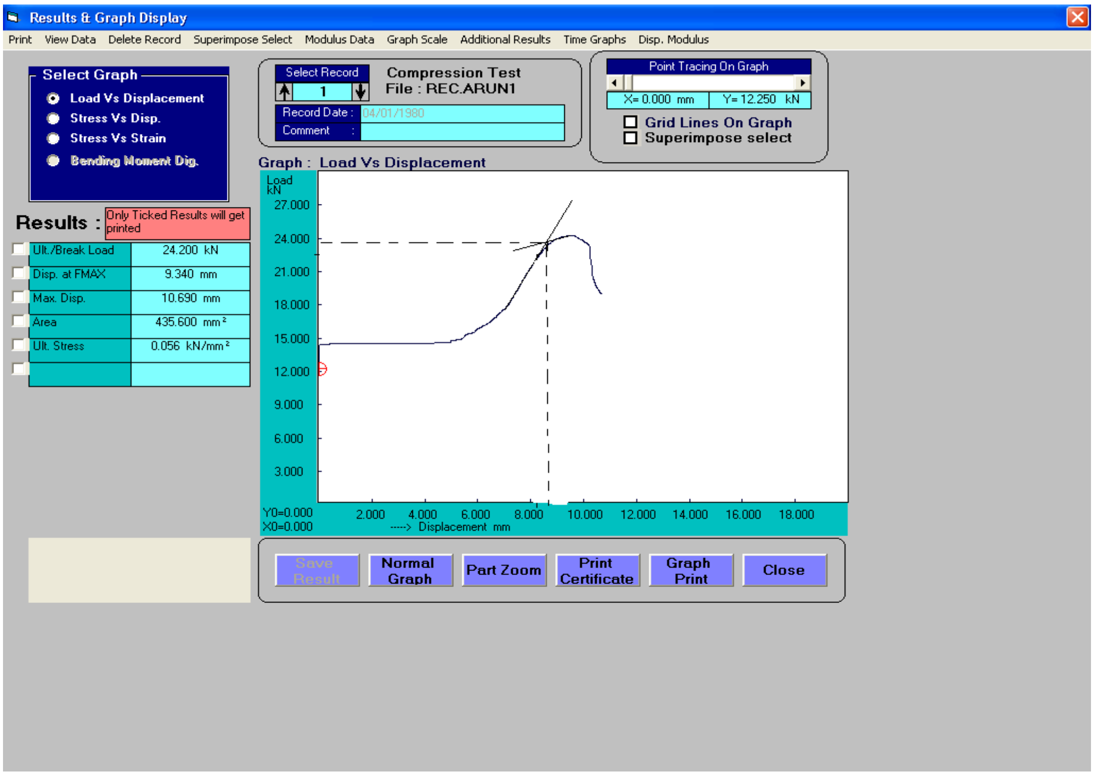


(Fig.33) Load v/s end shortening

The load v/s out of plane displacement and load v/s end shortening graph for plate 13 with rectangular cut out is shown in the following figure. It was observed that buckling load for this plate less as compared to the plate with circular cutout.

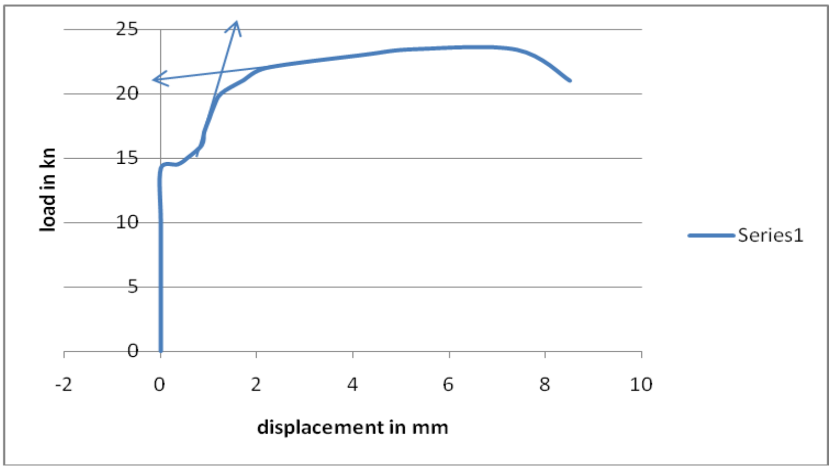


(fig.34) Load v/s out of plane displacement

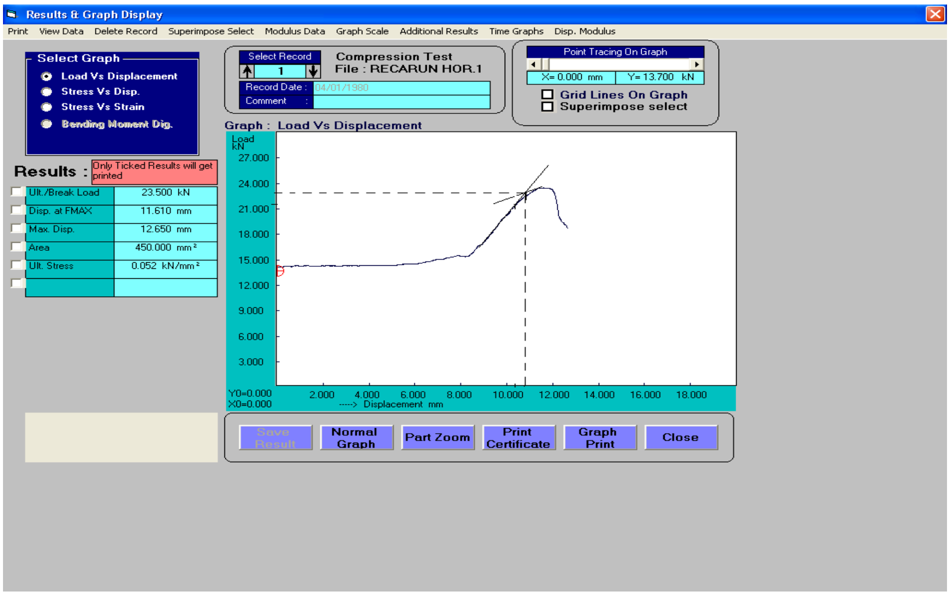


(fig.35) Load v/s end shortening

Plate-14 (12 layer,130*120*3.68mm , $[0/0/0/0]_{12}$) having the rectangular cutout in the transverse direction was subjected to compressive loading. The load v/s out of plane displacement and load v/s end shortening was plotted and is shown in fig.36 and 37 respectively. It was observed that the plate 14 gives less buckling load as compared to other plates with cutout.



(Fig.36) Load v/s out of plane displacement



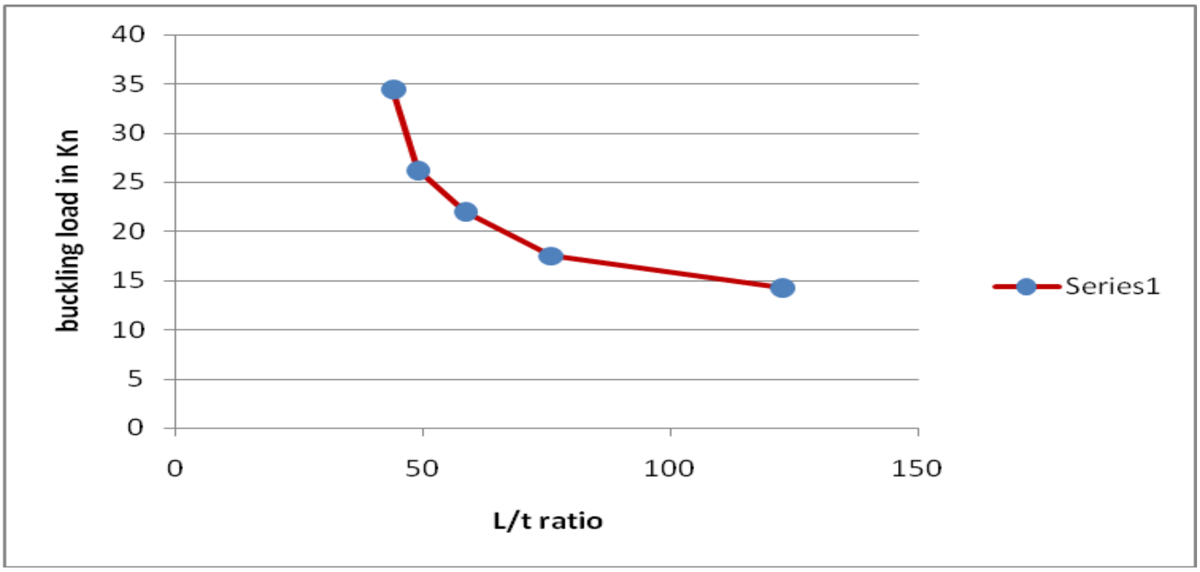
(Fig.37) Load v/s end shortening

Effect of Length to Thickness Ratio (L/t)

Plates with different thickness are used extensively due to design requirements. Thus, the buckling response of plates with its length to thickness ratio must be fully understood in the structural design. In this study the thickness of the plate was increased by increasing number of layers. The experimental results shows that the variation in buckling load is very sensitive to the thickness of the plate. The variation in buckling load with change in the thickness of the plate is shown in fig.38. From the graph in it is observed that the buckling load decreases with increase in length to thickness ratio.

Sl. no	Length mm	Width mm	Thickness mm	Buckling Load(KN) Graph1	Buckling Load(KN) Graph2
1	130	120	1.5	15.3	15.5
2	130	120	1.32	15.2	15.5
3	130	120	1.33	15.1	15
4	130	120	2.25	17.65	17.8
5	130	120	2.23	17.3	17
6	130	120	3.29	20	19
7	130	120	3.37	22.5	23
8	130	120	3.31	22	22.5
9	130	120	3.62	26.5	26
10	130	120	3.65	27	26.5
11	130	120	3.85	33.5	34.5
12	130	120	3.85	34	36

Table 3: Effect of Length to Thickness Ratio (L/t)



(Fig.38) Buckling load v/s length to thickness ratio.

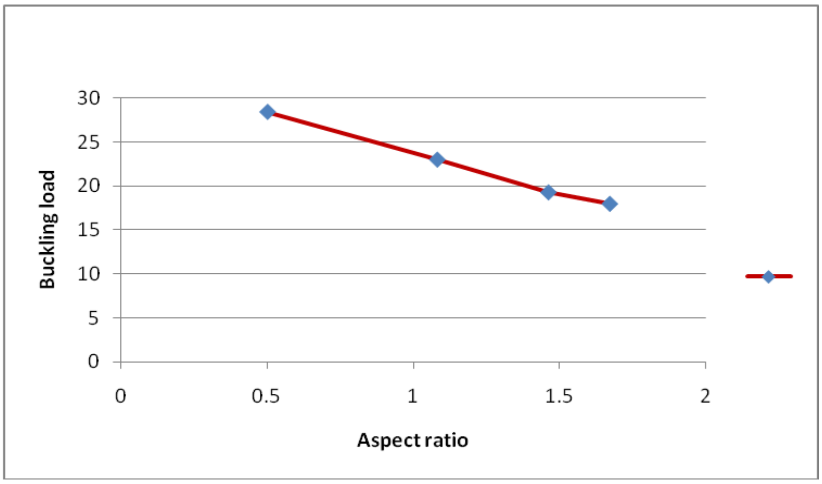
Effect of aspect ratio (a/b ratio)

In this study, the laminated plates are evaluated at four different aspect ratios. The tested plate and their corresponding buckling load was shown in table-4.The buckling load decreases continuously with increasing aspect ratio but the rate of decrease is not uniform. In this study aspect ratio was changed from 0.5 to 1.67.It is observed that buckling load was maximum for aspect ratio 0.5 and minimum for aspect ratio 1.67. When the aspect ratio changed from 0.5 to 1, the variation in buckling load is almost 24%. There are loss of 21% of buckling load between aspect ratios 1 and 1.5. The aspect ratio and buckling load was plotted along x and y axis as shown in fig.39. From that graph, it is observed that the rate of decrease in buckling load is decreasing with increase in aspect ratio.

Sl. no	Length mm	Width mm	Thickness mm	Buckling Load(KN) Graph1	Buckling load(KN) Graph2
1	130	120	3.29	20	19.8
2	130	120	3.37	22.5	23
3	130	120	3.31	22	23.5

4	175	120	3.31	18.2	18.7
5	175	120	3.2	17.9	18.4
6	175	120	3.15	18.9	19
7	200	120	3.2	17	18
8	200	120	3.1	17.5	18.2
9	60	120	3.29		27
10	60	120	3.2		26

Table 4 : Effect of aspect ratio (a/b ratio)



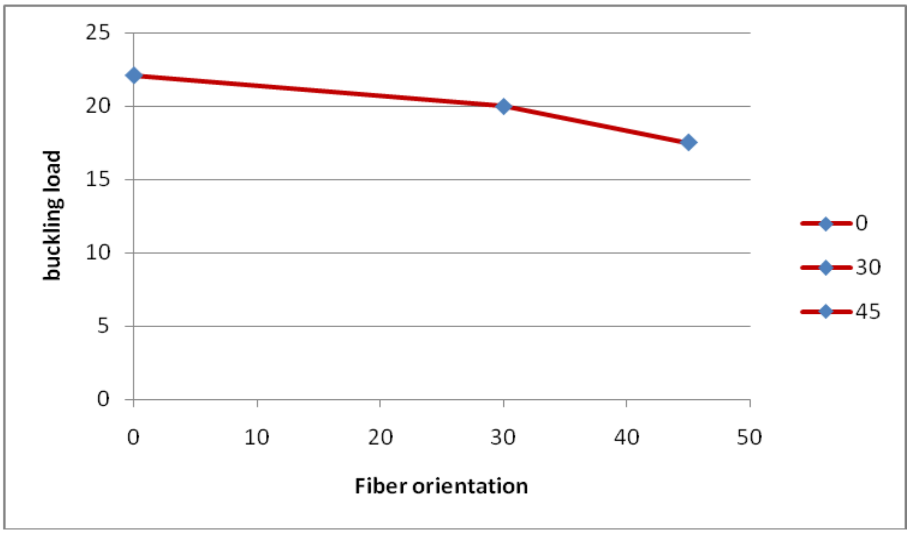
(fig.39) Buckling load v/s aspect ratio graph.

Effect of fibre orientation

In this study the buckling load of composite plates with different fiber orientation was determined. The result is shown in the Table-5 . The result shows that the buckling load is decreasing with increase in fiber orientation angle. The maximum buckling load was occurred at $[0]_8$. When the orientation of the fiber changed from 0^0 to 30^0 , the corresponding change in buckling load was almost 20% . Reduction of 30% in buckling load is observed as the ply orientation angle increases from 0^0 to 45^0. The variation of buckling load with fiber orientation shown in fig.40.

Sl. no.	orientation	Length (mm)	Width (mm)	Thickness (mm)	Buckling load(kn) From graph1	Buckling load(kn) from graph 2
1	0	130	120	2.96	22.5	22
2	0	130	120	3.3	22	23.5
3	0	130	120	3.37	23	23.5
4	30	130	120	3.31	21.05	20.5
5	30	130	120	3.28	20.5	20
6	30	130	120	3.25	20.0	20.5
7	45	130	120	3.18	16.3	16
8	45	130	120	3.24	17.2	17.5

Table 5 : Effect of Orientation



(Fig.40) Buckling load v/s fiber orientation

Effect of cut out shape

Plates with different types of cut outs are used extensively due to design requirements. Thus, the buckling response of plates with cut out must be fully understood in the structural design. In this section, the effects of circular, square and rectangular shapes with same areas are taken in to account. The experiments indicate that the variation of the buckling loads is very sensitive to the presence of cut out. It can be seen that buckling

load generally decreases with presence of cutout. We can observe that the buckling load for plate without cutout are about 25% and 30% higher than that of $[0]_{12}$ with circular and square cutout. The plate with rectangular cutout gives the minimum buckling load.

Sl. no	No. Of layers	Cutout shape	Length (mm)	Breadth (mm)	Thickness (mm)	Buckling Load(kn) Graph1	Buckling load(kn) Graph2
1	12	without	130	120	3.7	34.5	36
2	12	Circular	130	120	3.78	24.5	25.75
3	12	Square	130	120	3.81	23.75	24.5
4	12	Rectangular1	130	120	3.65	22	22.5
5	12	Rectangular2	130	120	3.7	23	23.8

Table 6 : Effect of cut out shape

CHAPTER-5

CONCLUSION

CONCLUSION

This study considers the buckling response of laminated rectangular plates with clamped-free boundary conditions. The laminated composite plates have varying L/T ratio, aspect ratio, cut out shape and ply orientation. From the present analytical and experimental study, the following conclusions can be made.

1. It was noted that different length to thickness ratio affected the critical buckling load. The buckling load decreases as the L/t ratio increases. The rate of decrease of buckling load is not uniform with the rate of increase of L/t ratio.

2. As the aspect ratio increases, the critical buckling load of the plate decreases. When the aspect ratio changed from 0.5 to 1, the variation in buckling load is almost 24%. The rate of change of buckling load with the aspect ratio is almost uniform.

3. It was seen that the different fiber orientation angles affected the critical buckling load. When the fiber angle increases, the buckling load decreases. The plate with $[0]_8$ layup has the highest buckling load and the plate with $[45]_8$ layup has the lowest buckling load.

4. The reduction of the buckling load due to the presence of a cutout is found to be significant. It is noted that the presence of cutout lowers the buckling load and it varies with the cutout shape. The plate with circular cutout yielded the greatest critical buckling load.

Future scope of the work

In the present study the buckling load of the laminated plate was determined. The effect of cutout shape, length to thickness ratio, aspect ratio and fiber orientation on buckling load was studied. The future scope of the present investigation can be expressed as follows,

(a) Buckling analysis of delaminted industry driven woven composite plates with and without cutouts.

(b) Buckling analysis of laminated woven fiber composite plates with delamination by numerical approach for different boundary conditions.

(c). Dynamic stability of woven fiber laminated and delaminated composite plates.

CHAPTER-6

REFERENCES

REFERENCES

1. A K Sreevastva, R.K Singh. Effect of aspect ratio on buckling of composite plates-. *Journal of Composites Science and Technology* 59 (1999) 439-445

2. Buket Okutan Baba and Aysun Baltaci. Buckling characteristics of symmetrically and anti-symmetrically laminated composite plates with central cutout,- *Applayed Composite Materials* – 14(2007):265–276

3. C.W. Pein and R. Zahari. Experimental investigation of the damage behaviour of woven fabric glass/epoxy laminated plates with circular cut-outs subjected to compressive force,- *International Journal of Engineering and Technology*, (2007)Vol. 4, No. 2, pp.260-265

4. Chavanan supasak, A comparison of experimental buckling load of rectangular plates determined from various measurement method-.*Department of mechanical engineering,Bangok*,(2006)18-20,2547

5. Chainarin Pannok and Pairod Singhatanadgid . Buckling analysis of composite laminate rectangular and skew plates with various edge support conditions. **-** *The 20th Conference of Mechanical Engineering Network of Thailand* (2006)18-20.

6..David Roylance, Laminated composite plates, Massachusetts Institute of Technology Cambridge, (2000) MA 02139.

7. D. Bucco and J. Mazumdar, Buckling analysis of plates of arbitrary shape, *journal of Austral. Math. Soc*. Ser. B 26 (1984), 77-91

8. E. A. Pieczyska, R. B. PecherskI and S.P. Gadaj . Experimental and theoretical investigations of glass-fibre reinforced composite subjected to uniaxial compression for a wide spectrum of strain rates. *Arch. Mech.,* 58 (2006), 3, pp. 273–291.

9. G. Garya and H. Zhao. Dynamic testing of fibre polymer matrix composite plates under in-plane compression- *Journal of Composites*: Part A 31 (2000) 835–840.

10 Ionel Chirica, Elena-Felicia and R. Chirica. Numerical tests on the buckling of the plates made of composite materials, *Romanian technical sciences academy*, vol.3(2006)

11. John Tomblin & Ever Barbero, Local buckling experiments on FRP Columns, *Journal of Thin-Walled Structures* 18 (1994) 97-116.

12. K. K. Shukla; Y. Nath; E. Kreuzer; and K. V. Sateesh Kumar. Buckling of Laminated Composite Rectangular Plates , *Journal of aerospace engineering*, vol.18(2005):215.

13.K. M. Jeong and H. G. Beom, Buckling Analysis of an Orthotropic Layer Bonded to a Substrate with an Interface Crack, *Journal of composite materials*, Vol. 37, No. 18(2003).

14. M. Kamruzzaman, A. Umar and S. Q. A. Naqvi. Effect of composite type and its configuration on buckling strength of thin laminated composite plates, *Latin American Journal of Solids and Structures* 3 (2006) 279-299.

15. M.R. Bambach, Local buckling and post-local buckling redistribution of stress in slender plates and sections, *Journal of Thin-Walled Structures* 44 (2006) 1118–1128.

16. M.R. Khalili, K. Malekzadeh, R.K. Mittal, A new approach to static and dynamic analysis of composite plates with different boundary conditions, *Journal of Composite Structures* 69 (2005) 149–155.

17. M. Darvizeh, A. Darvizeh , R. Ansari and C.B. Sharma. Buckling analysis of generally laminated composite plates- *journal of composite structures* ,vol.63(2004), pages 69-74.

18. Murat Yazic, Buckling of Square Perforated Thermoplastic Composite Plates, *Journal of Reinforced Plastics and Composites* (2008) 27; 1059.

19. N G R Iyengar . Structural stability of columns and plates, *EWP pvt. Ltd*.

20. Qing-Qing Ni , Jia Xie and Masaharu Iwamoto, Buckling analysis of laminated composite plates with arbitrary edge supports, *Composite Structures* 69 (2005) 209–217.

21. Rizal Zahari , Abdul Hannan Azmee and Faizal Mustapha, Prediction of progressive failure in woven glass/epoxy composite laminated panels**,** *Jurnal Mekanikal* ,June 2008, No. 25, 80 - 91

22. Stephen p. Timoshenko and Kriger- Theory of plates and shells-*McGraw-Hill tnternational editions.*

23. Shun-Fa Hwang and Ching-Ping Mao, Failure of Delaminated Carbon/Epoxy Composite Plates under Compression- *Composites Science and Technology* 61 (2001),1513–1527

24. S. M. Ibrahim, N. A. Siddiqui and S. Q. A. Naqvi, A simple procedure for reliability assessment of thin composite plates against buckling, *Latin American Journal of Solids and Structures* 4 (2007) 267-297.

25. Timothy L. C. Chen and Charles W. Bert. Design of composite-material plates for maximum uniaxial compressive buckling load - *Proc. Okla. Acad. Sci*. (1976)56: 104-107.

26. Wen-pei, Cheng, Ming-hsiang and Cheer-germ, Analysis modeling for plate buckling load of vibration test, Journal of Zhejiang University science *2005 6A(2):132-140.*

27. Wu Zhen and Chen Wanji, Buckling analysis of angle-ply composite and sandwich plates by combination of geometric stiffness matrix, *Journal of Comput. Mech*. (2007) 39: 839–848.

28. X. Wang and G. Lu . Local buckling of composite laminar plates with various delaminated shapes – *Journal of Thin-Walled Structures* 41 (2003) 493–506.

29. Yeliz Pekbey. A Numerical and Experimental Investigation of Critical Buckling Load of Rectangular Laminated Composite Plates with Strip Delamination -. *Journal of Reinforced Plastics and Composites;* (2006)25; 685.